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FEASIBILITY STUDY OF MODERN AIRSHIPS

Phase II

VOLUME I - HEAVY LIFT AIRSHIP VEHICLE

BOOK I - OVERALL STUDY RESULTS

GOODYEAR AEROSPACE CORPORATION
AKRON, OHIO

SEPTEMBER 1976

CONTRACT NAS2-8643

PREPARED FOR AMES RESEARCH CENTER
MOFFETT FIELD, CALIFORNIA

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FEASIBILITY STUDY OF MODERN AIRSHIPS
(PHASE II)

Volume I - Heavy Lift Airship Vehicle

Book I - Overall Study Results

Contract NAS2-8643

September 1976

Prepared for

Ames Research Center, Moffett Field, California

by

Goodyear Aerospace Corporation, Akron, Ohio

FOREWORD

Goodyear Aerospace Corporation (GAC) under a jointly sponsored NASA/Navy Contract (NAS2-8643) has conducted a Phase II investigation into the feasibility of modern airships. The Ames Research Center and the Navy Air Development Center were the respective NASA/Navy sponsoring agencies. The Phase II investigation has involved further study of mission/vehicle combinations defined during the Phase I portion of the contract. NASA Contractor Report NASA CR-137692 summarizes the GAC Phase I investigation.

Volume I of the Phase II final report summarizes the work performed relative to a Heavy Lift Airship combining buoyant lift derived from a conventional helium filled airship hull with propulsive lift derived from conventional helicopter rotors. Contract funding for the effort reported in Volume I was $96,000.

Dr. Mark Ardema, the NASA Project Monitor, provided valuable technical guidance and direction to the entire study effort. Mr. Ralph Huston was the GAC Program Manager. Gerald Faurote was the Project Engineer for the Heavy Lift Airship investigation. Other principal personnel included:

Senior Technical Analyst	W. N. Brewer
Engineering Design	N. D. Brown
Control Systems Analyst	D. W. Lichty
Computer Analyst	N. P. Tomlinson

Subcontractors supporting the GAC study team included:

Aerodynamics/Stability & Control
Nielsen Engineering & Research

Institutional/Operational Constraints
Battelle Columbus Laboratories

Helicopter Performance/Operational Data
Piasecki Aircraft Corporation

Other contributors were:

CH-54 Weight, Cost, Performance, and Aerodynamic Characteristics; CH-54B Modification Guidance
Sikorsky Aircraft

Heavy-Lift-Helicopter Fly-By-Wire Technology
General Electric Corporation

Heavy-Lift Helicopter Precision Hover System Technology
Radio Corporation of America

The contractor wishes to acknowledge that NASA Ames Research Center (ARC) provided the use of the ARC 7 x 10-foot Wind Tunnel Facility for the purpose of an exploratory evaluation of the Phase II Heavy Lift Airship.

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NOMENCLATURE

Symbol	Description
AFCS	automatic flight control system
DOC	direct operating cost
DOF	degrees of freedom
FBW	fly-by-wire
FRV	flight research vehicle
GW	gross weight
FPS, fps	feet per second
Ft, Ft, ft	feet
Ft/Min, fpm	feet per minute
g	acceleration of gravity
HLA	heavy lift airship
HLH	heavy lift helicopter
HOGE	hover out of ground effect
HR, HRS	hour
HZ	Hertz ycles/second)
IFR	instrument flight rules
IGE	in ground effect
IOC	indirect operating cost
ISO	International Standards Organization
Kt(s)	knots
LTA	lighter than air
lb(s)	pounds
MPH, mph	miles per hour
Mi	statute miles

NOMENCLATURE (Continued)

Symbol	Description
n. mi. (N.M.)	nautical miles
OGE	out of ground effect
oz	ounces
PHS	precision hover sensor
PSI	pounds per square inch
SAS	stability augmentation system
S.L.	sea level
STOL	short takeoff and landing
TAS	true air speed
TOC	total operating cost
TOGW	takeoff gross weight
V	velocity
VFR	visual flight rules
VTOL	vertical takeoff and landing
$\forall$	envelope volume
X	horizontal position reference in inertial axis system
Y	horizontal position reference perpendicular to X in inertial axis system
yd(s)	yards
Z	vertical position reference in inertial axis system
α	angle of attack
β	angle of sideslip
θ	LTA pitch attitude

NOMENCLATURE (Concluded)

Symbol	Description
ϕ	LTA roll attitude
ψ	LTA yaw attitude
°	degrees (angle)
°F	temperature in degrees Fahrenheit
'	ft

FEASIBILITY STUDY OF MODERN AIRSHIPS
(PHASE II)

VOLUME I - HEAVY LIFT AIRSHIP VEHICLE

Goodyear Aerospace Corporation

1.0 SUMMARY

A Heavy Lift Airship (HLA) combining buoyant lift derived from a conventional helium-filled non-rigid airship hull with propulsive lift derived from conventional helicopter rotors has been investigated. The buoyant lift essentially offsets the empty weight of the vehicle; thus the rotor thrust is available for useful load and to maneuver and control the vehicle. Such a vehicle is capable of providing a quantum increase in current vertical lifting capability. In addition to this new dimension in unitary lift capability certain critical deficiencies of past airships are significantly minimized or eliminated.

The specific HLA configuration considered has a payload capacity of 68,040 Kg (75 tons) and a non-refueled range of 1.852×10^5 m (100 nautical miles). This payload capacity, which is in excess of six times that of the largest U. S. helicopter and in excess of four times that of the largest projected U. S. helicopter, is sufficiently large to transport a wide range of civil and military loads. Currently, military airborne heavy lift scenarios consider aggregate loads requiring payload capacities up to 126,980 kg (140 tons) while potential civil applications could involve several hundred tons.

The HLA concept, which is illustrated in Figure 1.1, combines four CH-54B helicopters by means of an interconnecting structure to a two and one half million cubic foot non-rigid airship hull.

The vehicle studied is sufficiently compact to fit within existing facilities and is controlled from the aft left helicopter by a command pilot by use of proven Fly-By-Wire (FWB) techniques. Automatic flight control and hover modes, with the hover capability enhanced by a Precision Hover Sensor (PHS), are provided in addition to the manual flight modes.

The helicopters are retained in their currently configured condition to the largest extent possible in order to minimize the cost of the first HLA vehicle. It is believed that an actual operational vehicle would utilize a central control car with only the main rotors or rotor/turbine modules retained on the outriggers. Such an operational configuration is illustrated in Figure 1.2. A study of the potential market size would be necessary first, however, to justify a sufficient quantity of vehicles over which to amortize the technology efforts needed to develop a refined configuration.

Various structural arrangements and material trade studies were performed in order to minimize structural weight, while maintaining acceptable acquisition costs. A reasonably detailed point design analysis was performed on the arrangement finally selected and as a result the empty weight estimates for the vehicle are believed to be very realistic. Figure 1.3 summarizes the basic characteristics of the Phase II HLA configuration.

A Six Degree-Of-Freedom (6 DOF) flight dynamics simulation, developed as a part of a corporately sponsored effort, served as the primary synthesis tool in the development of the HLA FWB control laws and autopilot systems. The Automatic Flight Control System (AFCS), PHS and FBW electronics of the HLA involve principles, techniques, and hardware developed and demonstrated during

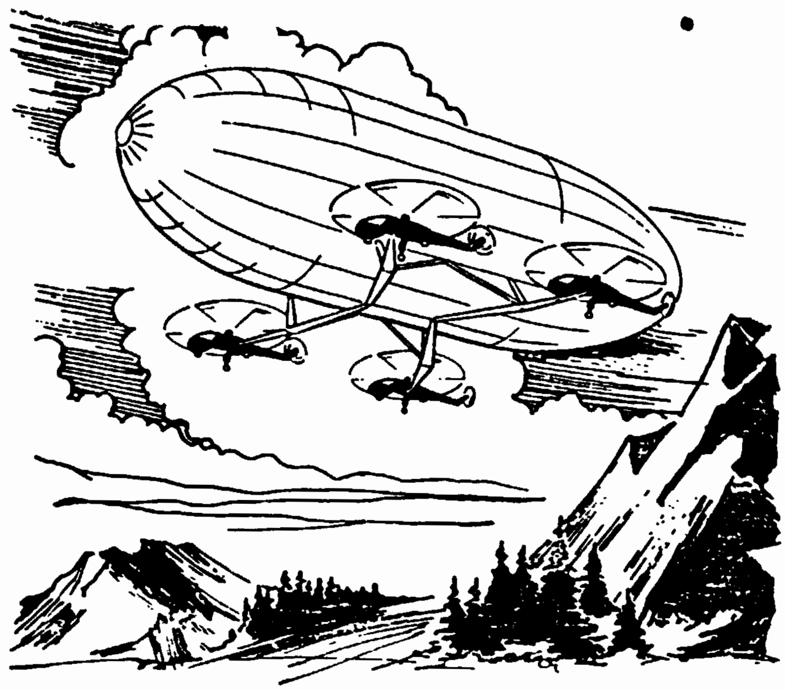


Figure 1.1 - Phase II Heavy Lift Airship Concept

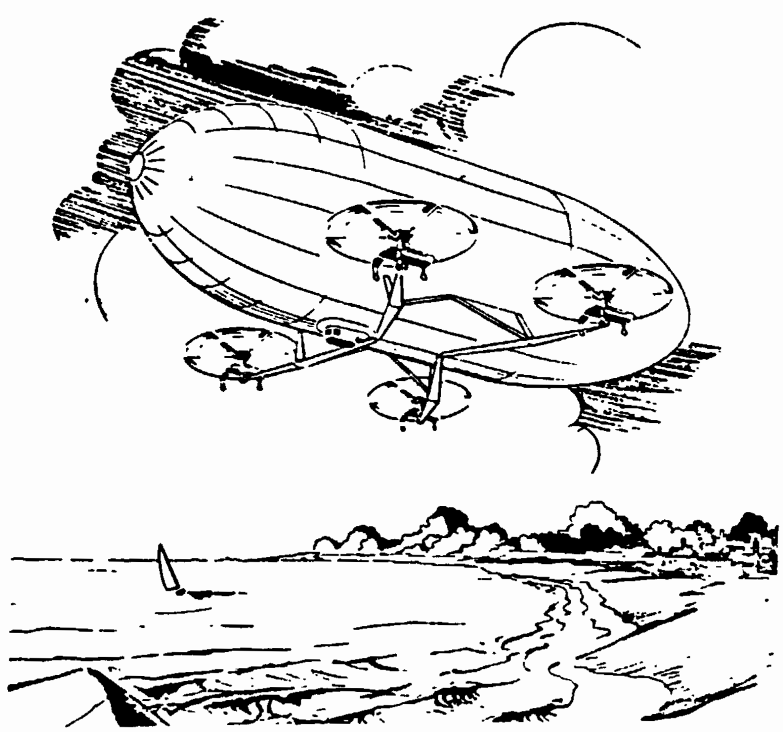


Figure 1.2 - Operational Heavy Lift Airship Concept

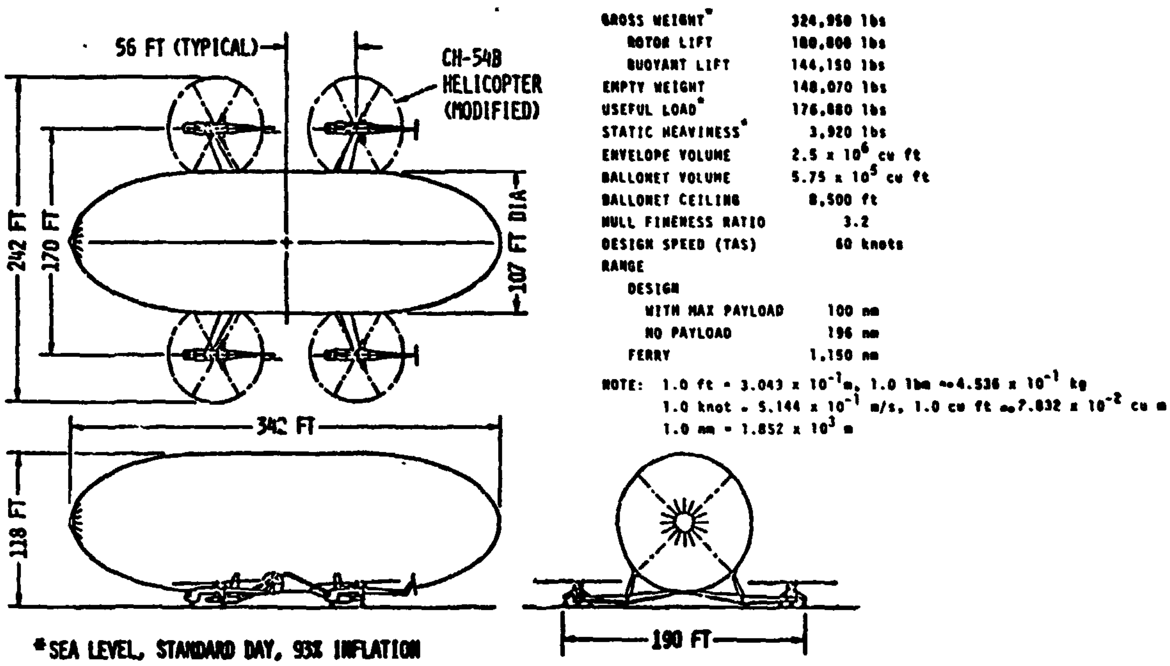


Figure 1.3 Phase II HLA Characteristics

the U. S. Army Heavy Lift Helicopter (HLH) program and a NASA-Langley program during which Sikorsky Aircraft modified a CH-54B helicopter to obtain a FBW capability.

A corporately funded powered wind tunnel model of the HLA was evaluated during the Phase II effort in the NASA Ames 7 x 10-foot wind tunnel facility. The results of these tests indicate the feasibility of combining large rotors in close proximity to a large hull. These tests have also indicated that the cross-wind station keeping capability of the vehicle can be noticeably improved through modifications to the current HLA configurations.

The Total Operating Costs (TOC) of the HLA on a payload ton-mile basis are shown to be substantially reduced over current large helicopter vertical lift costs basically due to the economic leverage afforded by the buoyant lift. Given proper technology programs in the area of low maintenance rotor concepts, the TOC for the HLA will become more favorable than defined herein. In economic comparisons involving payload weights beyond the capability of current helicopters, the cost of the alternative to the HLA must include: construction of special roads or port facilities; in-field assembly when large items must be shipped on a component basis to the point of use; loss of efficiency due to the size limitations that the alternative transportation modes impose, etc.

The Technology Assessment Analysis has indicated that a Flight Research Vehicle (FRV) is required to support the acquisition of technical information needed in the development of HLA vehicles meeting current and projected civil and military heavy lift needs. Such a vehicle is a requirement to obtain research capabilities that cannot be duplicated in ground-based facilities or in ground-based component and subsystem testing. In addition, this vehicle will:

(1) serve a concept verification function,

(2) provide a means to illustrate advances afforded by new technology,

(3) serve to establish potential user confidence, and

(4) illustrate economic competitiveness.

The contractual and corporate technical efforts, which included evaluation of a powered wind tunnel model and the development of a 6 DOF flight dynamics hybrid computer simulation, indicate the basic feasibility of the HLA concept. The development plan recommended herein is believed adequate to insure the successful development of the needed FRV. Goodyear recommends that the Phase II HLA configuration be considered as a point of departure in the development of the FRV. The research vehicle could be completed within 44 months and available for research and proof-of-concept flights within 50 months.

The recommended FRV would maximize use of existing government assets consistent with the needed research capabilities.

The major existing assets that can be utilized and that have been considered during the Phase II effort include the:

(1) CH-54B helicopters

(2) HLH PHS

(3) HLH AFCS

(4) HLH cargo handling system

(5) NASA Piloted Aircraft Data System (PADS) flight research instrumentation system

(6) Air pressure system components (damper valves, air valves, helium valves, etc.) currently in storage at NAF Lakehurst

The ability to utilize a substantial number of existing assets results in a significant reduction in the cost of the research vehicle without appreciably diminishing the research worth.

2.0 INTRODUCTION

2.1 Background

During Phase I of the Modern Airship Study, Goodyear identified a transportation mission involving the short haul of heavy outsized cargo well beyond current helicopter capacities. Various vehicle concepts (References 1, 2, 3, and 4) combining buoyant and rotor lift have been proposed for performing these emerging heavy lift short haul missions. Goodyear's Phase I review of these vehicle concepts, identified the approach proposed by Piasecki Aircraft Corporation (Reference 1) as having several basic benefits deserving of further study during Phase II of the Modern Airship Study. The basic benefits postulated for the concept are:

1) Minimization or Elimination of Prior ITA and Helicopter Deficiencies including:

a) LTA

(1) Elimination of ballast/payload interchange

(2) Low speed control

(3) Hovering

(4) Ground handling

b) Helicopter

(1) Fuel consumption

(2) Airframe weight

(3) High maintenance costs

2) Reduction in the Present Cost of Vertical Lift

3) Minimal RDT&E Costs Due to:

(1) Simplicity of the concept

(2) Use of proven, certified helicopter components

(3) Use of proven LTA concepts and materials

(4) Use of LTA air pressure system components currently in storage at NAF Lakehurst

(5) Use of fly-by-wire concepts developed during the HLH program for control of tandem main rotors

(6) Use of the PHS, AFCS and cargo handling system hardware developed during the HLH Program.

(7) Prior conversion of the CH-54B (by Sikorsky for NASA-LRC) to a fly-by-wire capability

(8) Use of NASA PADS flight research instrumentation system

The Phase II concept is relatively immune to scale effects and prior analysis (Reference 5) has shown that useful loads up to several hundred tons appear practical based upon the use of existing rotor systems.

2.2 Scope of Phase II Investigation

The various statement of work tasks and study sub-tasks and their interrelation is illustrated in Figure 2.1. The study consisted of four individual tasks as indicated. Task I, which involved the definition of the vehicle design characteristics, was a major area of concentration during the study effort.

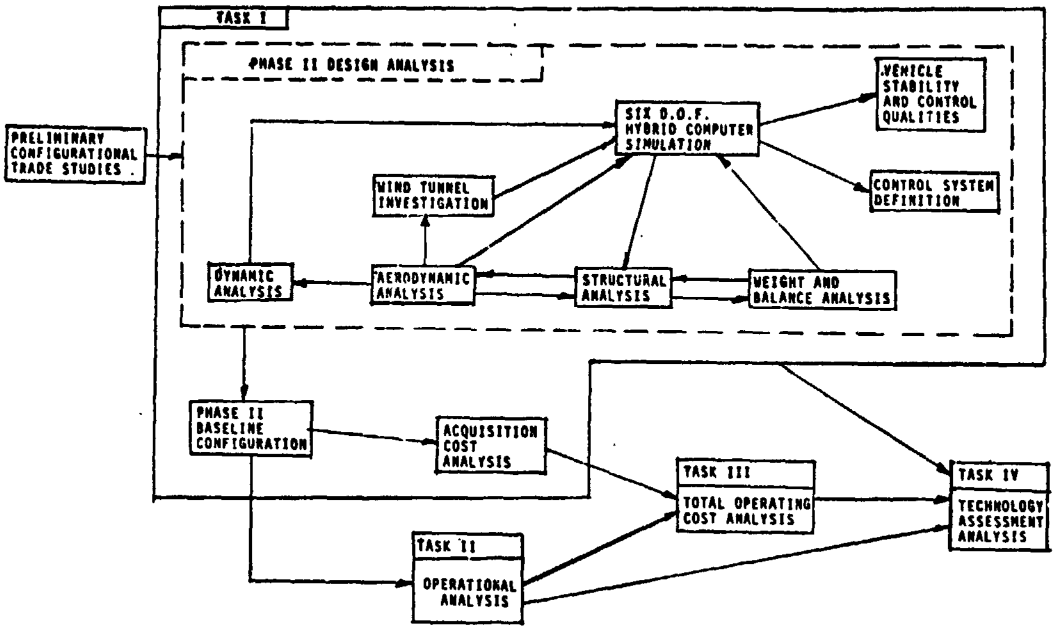


Figure 2.1 Phase II HLA Study Methodology

Corporately funded investigations beyond the contractual scope were undertaken to secure the best possible assessment of the technical questions that normally surround any new airborne vehicle concept. Corporately sponsored activities included the development of a six degree of freedom (6 DOF) hybrid computer simulation and the design, analysis, and fabrication and testing of a powered wind tunnel model. A contract modification was received near the end of the program providing government support for the analysis and application of the wind tunnel data. The wind tunnel tests were conducted in the NASA Ames 7 x 10-foot facility.

The Phase II design analysis effort, which is enclosed within the dashed lines of Figure 2.1, was a highly iterative process between the various elements shown. As indicated in Figure 2.1, once the Phase II baseline configuration was established, a vehicle acquisition cost analysis effort was initiated. Acquisition costs were estimated as a function of fleet size with special manufacturing facilities expenses accounted for. The acquisition cost analysis was based upon the projected nature of the operational configuration illustrated in Figure 1.2.

Task II consisted of the definition of operating procedures for the HLA. Included were the definition of: flight and ground crew requirements; maintenance concepts, in-flight operating procedures, ground handling procedures and payload management procedures. The effects of physical and institutional constraints on the vehicle's operation were identified.

Task III consisted of the development of the direct and indirect operating costs for the HLA in a commercial operations scenario. The resulting total operating costs were then compared with those of the largest current helicopters to secure an assessment of the cost competitiveness of the HLA.

Based on Tasks I, II, and III, a Technology Assessment Analysis was conducted which included the following major subtasks:

1) Identification of important technology areas where substantial contributions toward safety, economics, or performance could be achieved.

2) Identification of the need for flight research vehicles.

3) Identification of development costs and schedule.

3.0 HEAVY LIFT MISSION RATIONALE

3.1 General

The shipping, railway, trucking, and CTOL aircraft industry generally meet the demands of world commerce with respect to the long-haul of conventional cargo with the exception of underdeveloped areas. The helicopter has provided a short-haul vertical lift capability; however, as is generally well known, the growth of the helicopter payload capacity has been very slow. This trend will continue due to scale effects associated with the helicopter class of vehicles. The largest current inventory helicopter in the U. S. is the Sikorsky S-64F/CH-54B crane which is rated at 11,340 kg (12.5 tons) at sea level with a range of 74,080 m (40 nautical miles) at 288 deg K (59 deg F). Since the demise of the HLH program, which was oriented to the development of a payload capacity on the order of 27,216 kg (30 tons), there is currently no short haul vertical lift capability projected beyond the Sikorsky CH-53E helicopter, which when operational will have approximately a 16,329 kg (18 ton) payload capability.

3.2 Commercial Applications

There is a growing national need for heavy lift capacities far beyond 16,329 kg (18 tons). It appears that this market would grow quickly if an economical heavy lift transportation system can be provided. The economic analysis of Section 9.0 of this report indicates that the TOC per ton mile for the HLA are substantially reduced over current large helicopter vertical lift costs. Thus, it can also be anticipated that the HLA would capture portions of existing helicopter markets.

The commercial heavy lift missions include items that are oversize and/or overweight and cannot be transported over present roadways or railroads or the item represents an infrequent shipment to a region not otherwise requiring right-of-ways for highways or railroads. Commercial and institutional heavy lift missions are identified in Table 3.1 along with typical cargo weights and range requirements.

Power generating equipment is oversize and overweight for land shipment and represents the heaviest and most dense unit loads identified for the HLA. In the past, waterways and special rail cars have been used to transport assemblies and subassemblies to the site. The present desire to locate power stations away from waterways and population centers because of environmental or safety reasons eliminates direct water or rail transportation to the site and creates a need for a special one-time transportation system. The transport of large industrial equipment includes a

TABLE 3.1 UNIQUE MISSIONS AND VEHICLE REQUIREMENTS (HEAVY/OUTSIZE CARGO)

Missions	Vehicle Requirements									
	Performance				Cargo/Equipment Reqmts			Transport Effectiveness		
Heavy-Lift Outsize Transportation	Take Off	Speed KTS	Range N.M.	Endurance HRS	Size	Weight TONS	Environment	Schedule	Interfaces With	Competes With
COMMERCIAL										
Power Generating Equipment	VTOL STOL	5-50	20-50 50-200	3 6	Out-sized	50-500	Med. g	Flexible	Ship or Rail	Special Ground Vehicles (+) Right of Way Costs
Large Industrial Equipment	VTOL STOL	5-50	20-50 50-200	3 6	Out-sized	50-250	Med. g	Flexible	Ship or Rail	Special Ground Vehicles (+) Right of Way Costs
Mining Equipment (Remote Sites)	VTOL STOL	5-50	200-400 400-600 600-1000	16 24 40	Out-sized	50-100	Med. g	Flexible	Ship or Rail	Special Ground Vehicles (+) Right of Way Costs
Prefabricated Buildings	VTOL	5-50	20-50 50-200	3 6	Out-sized	25-100	Low g	Flexible	No other Form	Building at Site (+) private truck
Large Aerospace Vehicles	STOL	50-100	200-400 400-600 600-1000	8 12 20	Out-sized	25-100	Low g	Flexible	No other Form	Special Vehicles
Construction Services	VTOL	5-50	0-50	3	Out-sized	25-100	Med. g	Flexible	No other	Performs services existing ground cranes cannot
Timber Harvesting	VTOL	5-50	2.5-50	4	May be out-sized	25-100	Med. g	Flexible	Yarder Rail/ Truck	Helicopter Private truck
Offloading of Cargo Ships (Foreign Application)	VTOL	5-50	20-100	2	Out-sized	100-200	Med. g	Flexible	Ship or Rail	Special Ground Vehicles (+) Right of Way Costs
INSTITUTIONAL										
Coast Guard Aids to Navigation (ATN)	VTOL STOL	50-30	200-400	24	Out-sized	300	Low g	Some Flexibility	Ships	Ships, Aircraft
Coast Guard Marine and Environmental Protection (MEP)	VTOL	5-30	200-400	24	Out-sized	300	Low g	Some Flexibility	Ships	Ships, Aircraft

NOTE: 1.0 kt = 5.144×10^{-1} m/s, 1.0 nm = 1.852×10^{3} m, 1.0 ton = 907.2 kg broad range of items associated with the construction of refineries, chemical plants, pipelines, and manufacturing plants. Again, these plants are often located away from waterways and population centers for environmental and safety reasons and do not require movement of outside products once in operation. The transport of prefabricated buildings includes movement of oversized "prefabricated" homes, offices, and factory building units from the factory to the site. The shipments are essentially one time to a site even though many (such as homes) can be delivered to a relatively concentrated area. The prefabrication of buildings permits the economics of scale at the factory to be effected to offset the rising cost of labor. For this approach to proliferate, however, an economical method of transport must be available.

As noted in Table 3.1, the commercial off-loading of cargo ships is primarily a foreign mission in areas not possessing deep water ports. This lack of deep water ports is currently a critical problem and is projected to be so for decades to come.

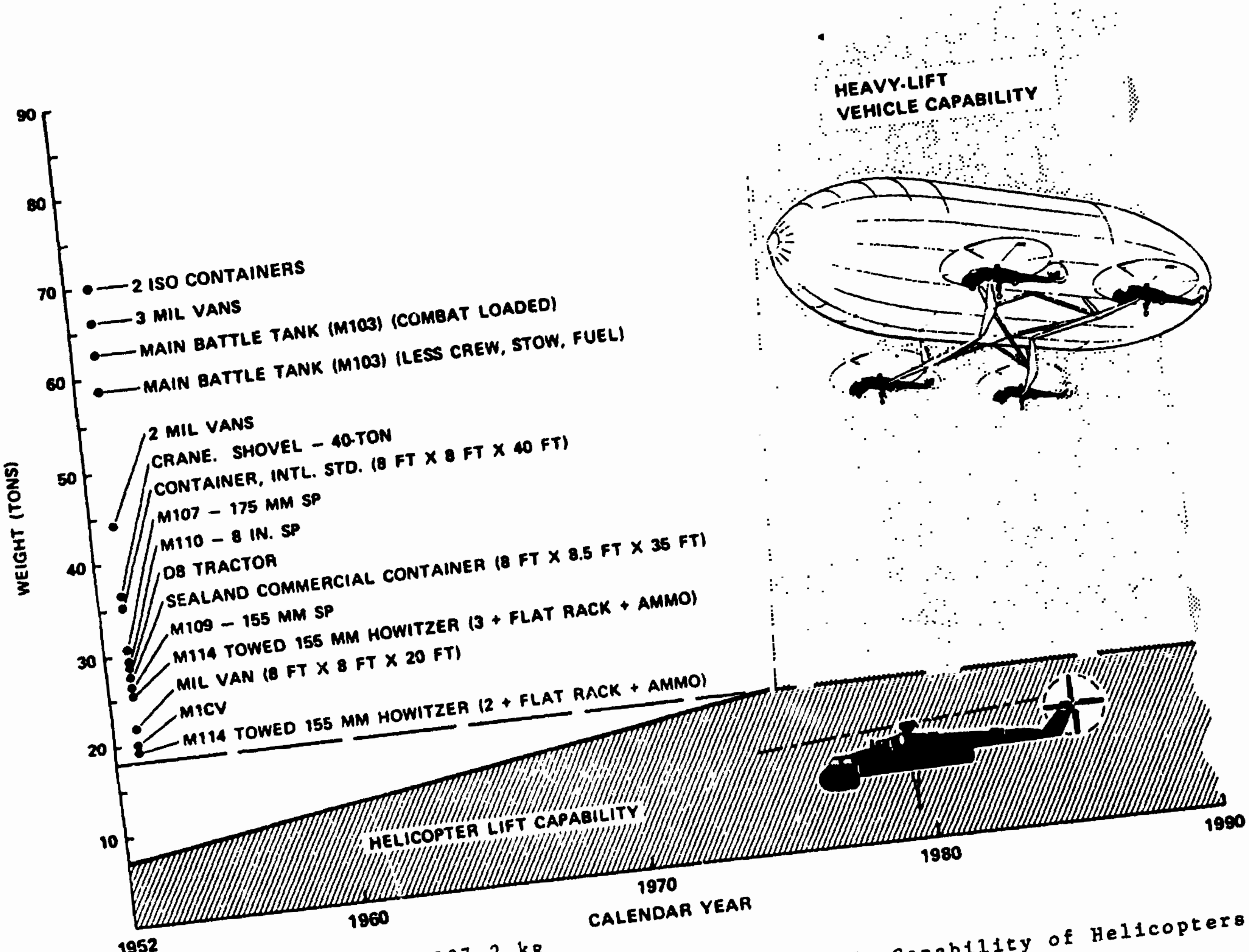
3.3 Military Applications

The off-loading of cargo from ships in areas lacking proper port facilities is a major military application of the HLA. This NAVAL mission, often referred to as the Logistic Over The Shore (LOTS) or Cargo (or Container) Over The Shore (COTS)[1] mission, involves the cargo shown in Figure 3.1 all of which is beyond current and projected helicopter capabilities. The flexibility and mobility afforded by the HLA class of vehicles offers a new dimension in the ability to quickly move massive quantities of material from any coastal area in the world to inland staging areas. The following is typical of the current LOTS mission rationale.[2]

---

[1] NAVAL COTS Program is Container Offloading and Transfer System

[2] The discussion refers to containerized cargo, however, not all over the shore cargo will be containerized and as indicated in Figure 3.1, single loads such as the fully loaded main battle tank require vertical lift capabilities up to 56,699 kg (62.5 tons).



NOTE: 1.0 ton = 907.2 kg

Figure 3.1 Military Cargo Beyond Lift Capability of Helicopters

3.3.1 LOTS Mission Rationale

The container ships are considered anchored off shore and accompanied by a tanker ship outfitted for refueling the heavy-lift vehicle. The distance between the ship and the shore base offloading site varies from one to ten nautical miles. The mobile equipment which handles the containers at the land base is capable of breaking down ganged containers stacked no higher than two. The width of the ganged container payload is not constrained by the handling equipment, but is limited by prompt access requirements to no more than two containers. Operation should be designed for performance of the mission in Sea State 5 and in crosswinds up to 15.43 m/s (30 knots). The vehicle must possess a ferry capability to the operational area and the environment is benign in terms of hostile enemy action.

3.3.1.1 Payload Weight

The maximum payload will be fixed at four (4) fully loaded ISO containers, grouped two high and two wide. Each container measures 2.44 m x 2.44 x x 12.19 m (8 ft x 8 ft x 40 ft) and weighs 31,751 kg (35 tons). The maximum payload weight is established at 127,006 kg (140 tons).[1]

3.3.1.2 Profile

1) Warmup and takeoff, Sea Level, 305°K (89°F), design TOGW

2) Cruise out from 1,852 m to 18,520 m (1 to 10 n.m.) at Sea Level, best endurance speed

3) Hover 2 minutes for offload

4) Cruise back at best endurance speed to starting point

5) Hover 2 minutes to reload

6) Repeat cycle until 20 minutes hover fuel remains

---

[1] This payload weight is essentially twice the capability of the vehicle investigated herein. A vehicle capable of meeting the 140 ton requirement is currently considered to be beyond the practical starting point for development of the concept.

3.3.2 Other Military Applications

Another military mission for which this class of vehicle has substantial promise and in which there has been recent interest is the movement of large completed missile assemblies and missile components. This and other Army and Air Force applications are listed in Table 3.2.

TABLE 3.2 OTHER MILITARY HEAVY-LIFT APPLICATIONS

Missions	Vehicle Requirements									
	Performance				Cargo/Equipment Reqmts			Transport Effectiveness		
	Take-Off or Mission	Speed (kts)	Range (nm)	Endurance (hrs)	Size	Weight (tons)	Environment	Schedule	Inter-faces With	Compares With
USAF										
Intra-Theatre Transport	STOL	50-100	200 - 1000	8-20	Out-size	25-100	Normal	All Weather	Surface	Limited Aircraft
Mobile ICBM Transporter	VTOL	5-75	200-400	4-10	Out-size	150-400	Normal	All Weather	None	Special Vehicles
U. S. ARMY										
Artillery Movement	VTOL	5-150	200-400	2-5	Out-size	0.5-15	Normal	All Weather	None	Helicopter
Large Load Lifter	VTOL	5-150	200-400	2-5	Out-size	25-50	Normal	All Weather	None	None
MBT/CEV* Lifter	VTOL	5-150	200-400	2-5	Out-size	50-75	Normal	All Weather	None	None

MBT - Main Battle Tank, CEV - Combat Engineer Vehicle

NOTE: 1.0 kt = 5.14×10^{-1} m/s, 1.0 ton = 907.2 kg, 1.0 nm = 1.852×10^{3} m

4.0 PHASE II VEHICLE AND MISSION REQUIREMENTS

The requirements considered in the Phase II HLA design analysis are:

Payload Weight (Maximum)	68,040 kg (75 tons)[1]
Adverse Weather	15.43 m/s (30 knots) crosswind
Precision Hovering Capability	Adequate to perform current helicopter vertical lift missions

[1] Sea level, standard day, power margin sufficient for 30.48 m/min (100 ft/min) vertical climb with one engine out

Maximum Still Air Speed With and Without Payload	30.86 m/s (60 knots) True Air Speed
Maximum Nonrefueled[1] Range with Maximum Payload	Not less than 1.852×10^5 m (100 nautical miles)
Ballonet Ceiling	2499 m (8200 ft)

While a vehicle meeting the above requirements falls far short of what appears to be the ultimate short haul heavy lift requirement per Section 3.0, it does represent a significant increase in current lift capability.

In addition, as discussed in Section 10 of this report, the Phase II requirements lead to a vehicle possessing sufficient research capabilities to permit acquisition of the full scale data needed to support the development of larger vehicles meeting projected civil and military needs.

5.0 DESIGN ANALYSIS

5.1 General

As indicated in Figure 2.1, the design analysis activity of Task I involved many elements of technical activity. Following several iterations between these elements a baseline configuration evolved as indicated in the figure. The design approach generally followed was to consider only proven hardware concepts, components, materials, manufacturing techniques, etc. This approach, if continued during a program to develop an initial test or research vehicle, can result in a low risk configuration and accordingly a high confidence development program. The development program recommended in the Technology Assessment Analysis (Section 10 of this report) includes an initial 24-month Final Systems Definition Phase which represents a continuation and significant expansion of the type of effort initiated during the Phase II design analysis task.

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1 Inflight refueling was accomplished in past airship operations and can be used to extend as necessary the range of the HLA.

It is believed that such a comprehensive initial phase is necessary if the research capabilities needed to develop HLA vehicles meeting current and projected needs are to be achieved.

5.2 Phase II HLA Configuration

The general arrangement of the HLA configuration that evolved from the Phase II design analysis is presented in Figure 5.1 (Goodyear Drawing 76-069). The vehicle is 104.24 m (342 feet) in length and the maximum diameter of the envelope is 32.61 m (107 feet). The overall height of the vehicle is 35.97 m (118 feet) with an overall width of 58.52 m (192 feet) thus the vehicle will fit within the type No. 3 sliding door hangars. There are on the order of fifteen hangars remaining in this country that could accommodate two such vehicles.

The Phase II configuration involves a two and one-half million cubic foot volume non-rigid hull fabricated from present day proven airship fabrics. The basic envelope and catenary curtain fabric is neoprene coated Dacron. The ballonet fabric is neoprene coated Nylon. Basic fabric and seam strengths required are only slightly greater than the maximum of the ZPG-3W airship built by Goodyear for the U. S. Navy in the late 1950's.

Twenty-three percent ballonets have been considered which result in a ballonet ceiling of approximately 2499 m (8200 feet) and an operational capability up to 1524 m (5000 feet) under all expected superheat conditions. For sea level operations a 93 percent envelope inflation would be used permitting a thousand feet of operational altitude and 11.1°K (20°F) of superheat.

The design gross weight[1] is 147,365 kg (324,950 lb) of which 65,372 kg (144,150 lb) is buoyant lift and 81,993 kg (180,800 lb) is rotor lift. As shown in Section 7.0 of this volume, the four helicopters are more than capable of providing this amount of lift with one engine out and adequate reserve for a 30.48 m/min (100 ft/min) climb.

[1]At sea level, standard day, 93% inflation.

The helicopters are attached to the buoyant hull by means of an interconnecting structure much of which is "submerged" within the envelope to reduce aerodynamic drag and overall vehicle height. Figure 5.2 illustrates the overall arrangement of the interconnecting structure and identifies the major subassemblies of the structure. The interconnecting structure consists of an internal star frame comprised of a series of welded three-boom girders with pin ended joints. The girders are fabricated from high performance steel, which after welding without subsequent heat treat, has 1.24×10^9 N/m^2 (180,000 psi) allowable stress level. Figures 5.3 through 5.7 (Goodyear Drawings 76-304; -305; -323; -330; and -332 respectively) provide the basic design details of the internal star frame.

The support and lift struts are of an aluminum honeycomb sandwich construction. Design details of these struts are provided in Figure 5.8 (Goodyear Drawing No. 76-082). The drag strut is a three-boomed girder of high performance steel similar in construction to the internal star frame girders.

Four modified Sikorsky CH-54B helicopters have been adapted to the interconnecting structure by means of a gimbal device. While substantial changes of direction in the main rotor thrust vector can be achieved by cyclic pitch control this approach cannot be used with the helicopters affixed rigidly to the interconnecting structure. With the helicopter rigidly affixed, large cyclic bending loads would be experienced in the main rotor mast which would unacceptably reduce the mast life. The gimbal permits the rotor mast to realign with the tilted thrust vector much the same as in normal helicopter flight. The helicopters are pitched about the gimbal by main rotor cyclic pitch and driven by servo controlled actuators in roll to negate gimbal coupling forces resulting from main rotor torque. Main rotor torque is counteracted by a differential cyclic pitch bias between port and starboard rotors. The bias is accomplished by an electrical input to the FBW flight control system. Thus, the tail rotors are not required for main rotor anti-torque purposes.

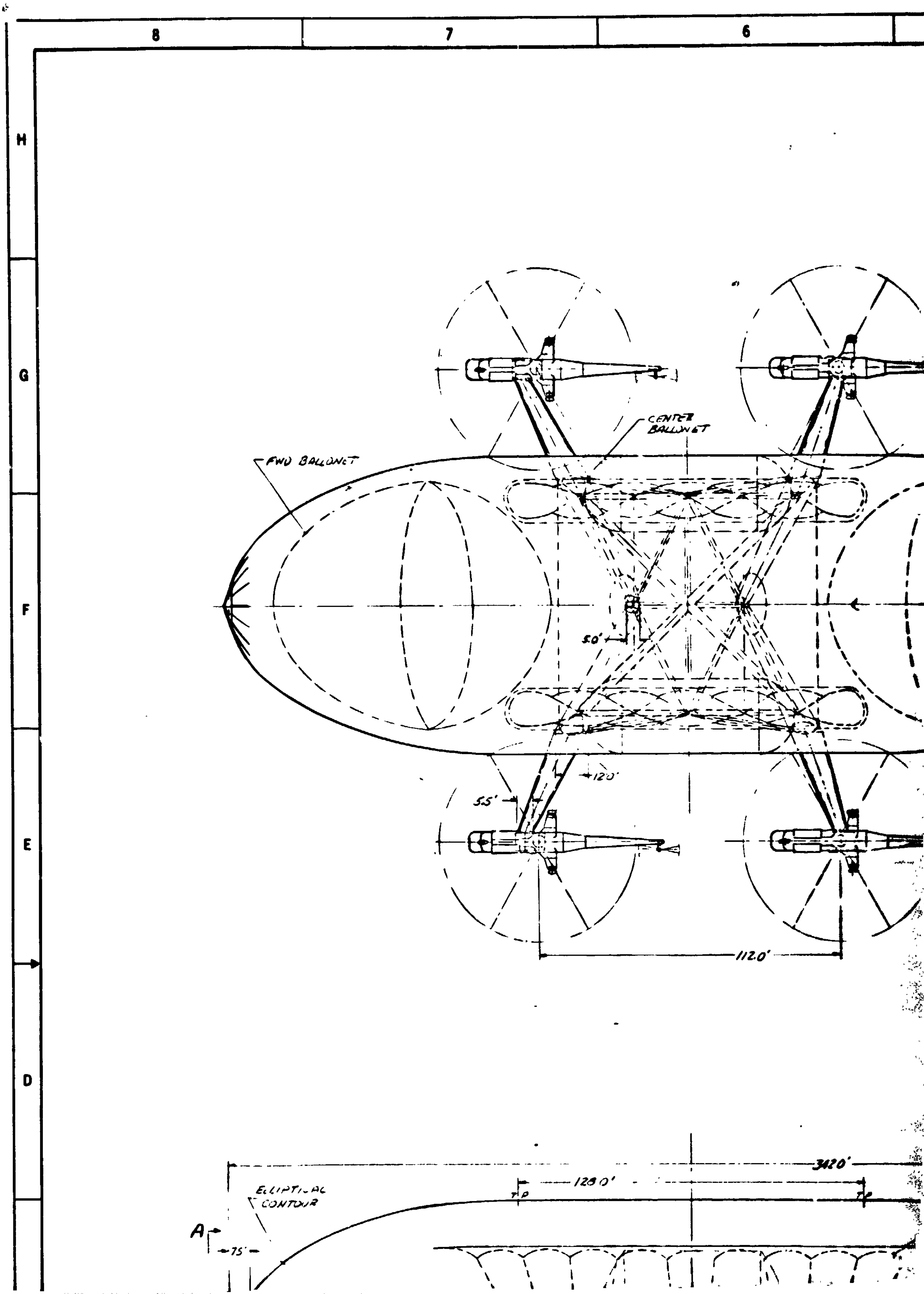
FIGURE 5.1 GENERAL ARRANGEMENT OF PHASE II HLA
(GOODYEAR DRAWING NO. 76-069)

NOTE: 1.0 ft = 3.048×10^{-1} m

1.0 cu ft = 2.832×10^{-2} cu m

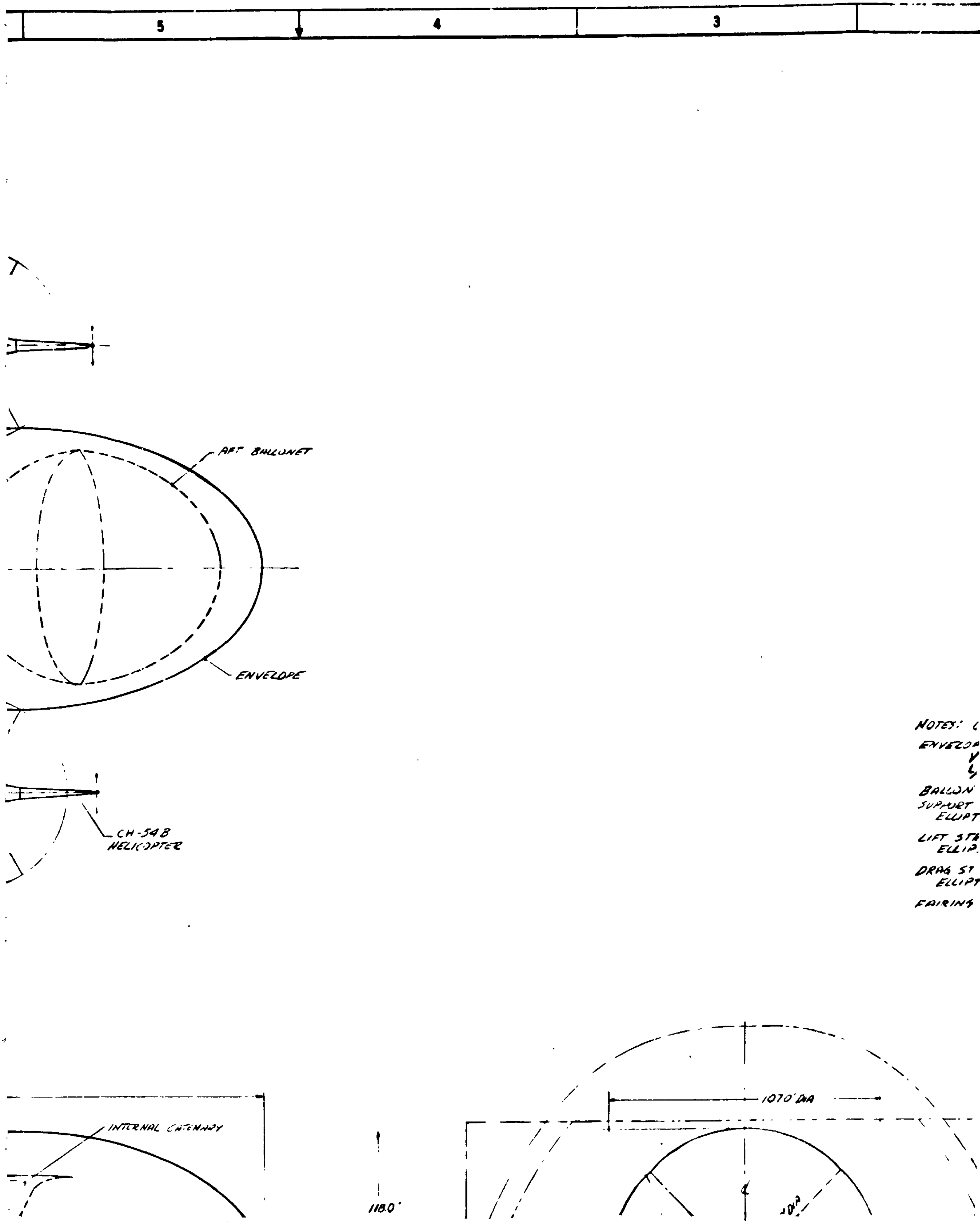
1.0 sq ft = 9.29×10^{-2} sq m

FOLDOUT FR



8
7
6
H
G
F
E
D
CENTER BALLONET
FWD BALLONET
5.0'
12.0'
5.5'
112.0'
342.0'
128.0'
ELLIPTICAL CONTOUR
TP
A
75'

FOLDOUT FRAME 2



3	2	1

REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED	
				BY	
				ENGR	
				CUST	

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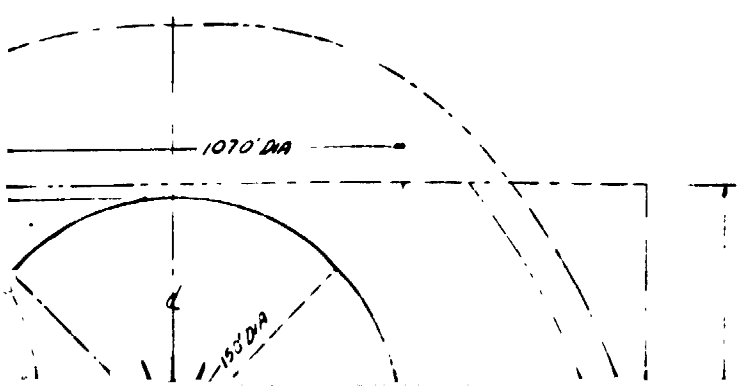
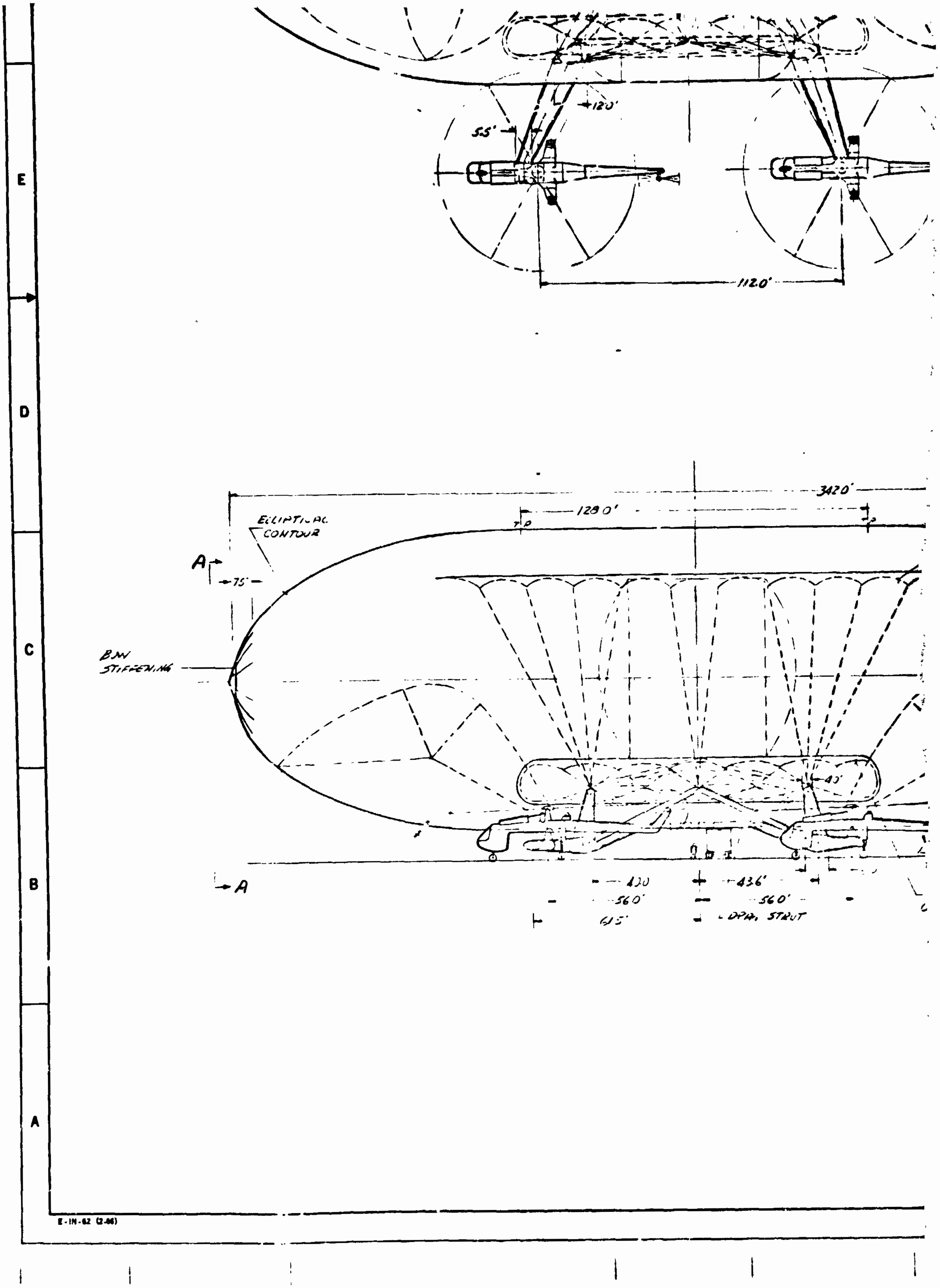
NOTES: UNLESS OTHERWISE SPECIFIED

ENVELOPE VOLUME	2,500,000 CU.FT.
$V^{2/3}$	18,297 SQ.FT.
L/D	3.2
BALLONET VOLUME	625,000 CU.FT.
SUPPORT STRUT PLAN FORM AREA	2922 SQ FT (TOTAL)
ELLIPTICAL CROSS SECTION T/C	.500
LIFT STRUT PLAN FORM AREA	432 SQ FT (TOTAL)
ELLIPTICAL CROSS SECTION T/C	.400
DRAG STRUT PLAN FORM AREA	344 SQ FT (TOTAL)
ELLIPTICAL CROSS SECTION T/C	.420
FAIRING FRONTAL AREA	77 SQ FT (TOTAL)

D





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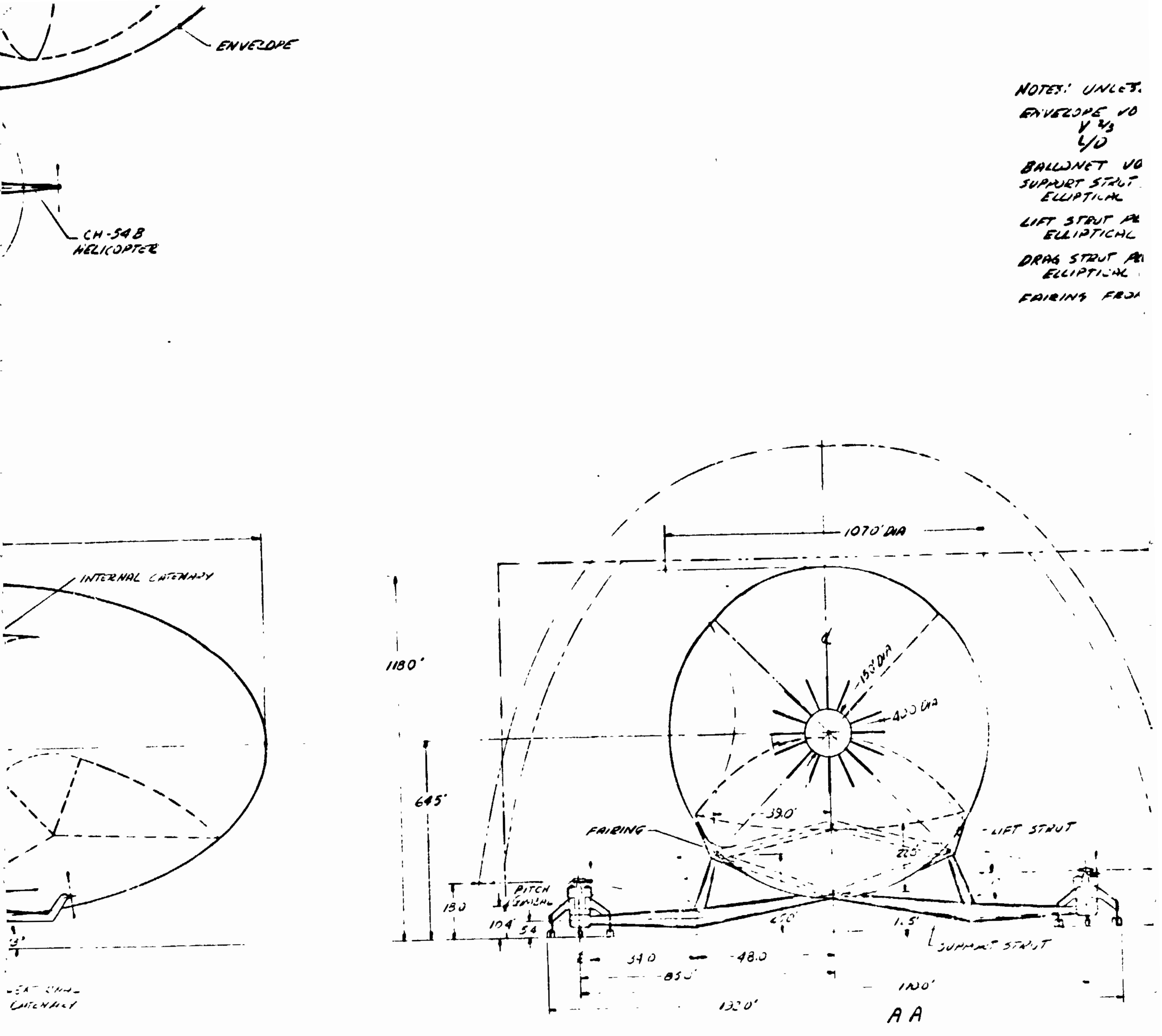
...OR QUALITY

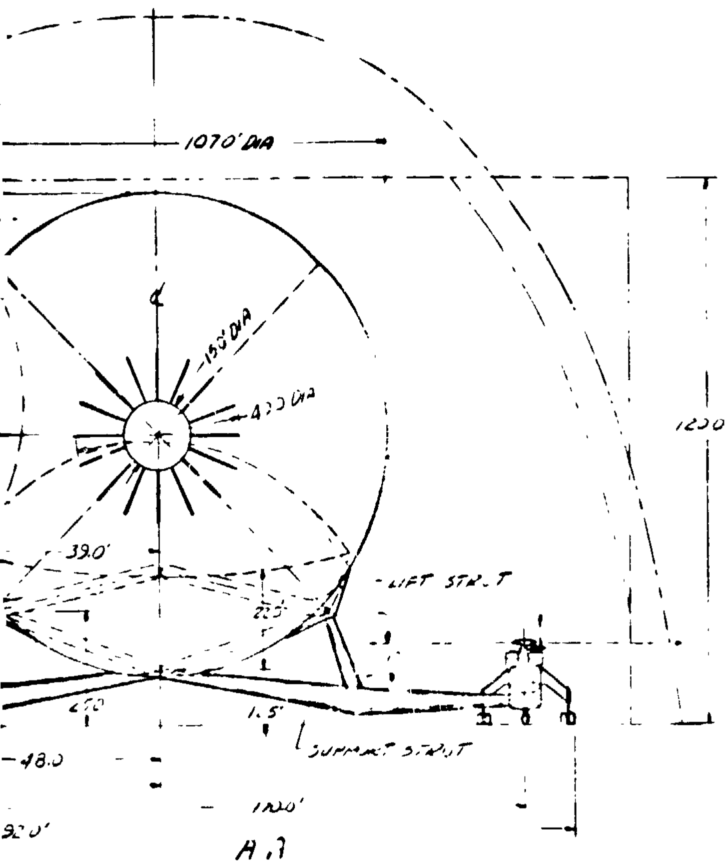


UNLESS OTHERWISE SPECIFIED
DIMENSIONS ARE IN INCHES
TOLERANCES ON:
DECIMALS .X ± .1
.XX ± .04
.XXX ± .010
ANGLES ± 1
MATERIAL
SEE DR. -9 60 FOR DRAWING
INTERPRETATION STANDARDS

FOLDOUT FRAME 5

NOTES: UNLESS OTHERWISE SPECIFIED

ENVELOPE VOLUME	2,500,000 CU.FT.
$V^{2/3}$	18,297 SQ.FT.
L/D	3.2
BALLONET VOLUME	625,000 CU.FT.
SUPPORT STRUT PLAN FORM AREA	3822 SQ FT (TOTAL)
ELLIPTICAL CROSS SECTION T/C	.500
LIFT STRUT PLAN FORM AREA	432 SQ FT (TOTAL)
ELLIPTICAL CROSS SECTION T/C	.400
DRAG STRUT PLAN FORM AREA	344 SQ FT (TOTAL)
ELLIPTICAL CROSS SECTION T/C	.420
FAIRING FRONTAL AREA	77 SQ FT (TOTAL)



UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES TOLERANCES ON DECIMALS X ± 1, .XX ± .04, .XXX ± .010, ANGLES ± 1°	CONTR / WTS / STRESS / ENGR / CHECK / DRAWN	GOODYEAR AEROSPACE CORP AKRON OHIO — HEAVY LIFTER
MATERIAL	GAC APPROVAL	SIZE E — CODE IDENT NO 25500 — DRAWING NO
SEE GRS-9 60 FOR DRAWING INTERPRETATION STANDARDS	CUSTOMER APPROVAL	SCALE — SHEET 1 OF

FOLDOUT FRAME 6

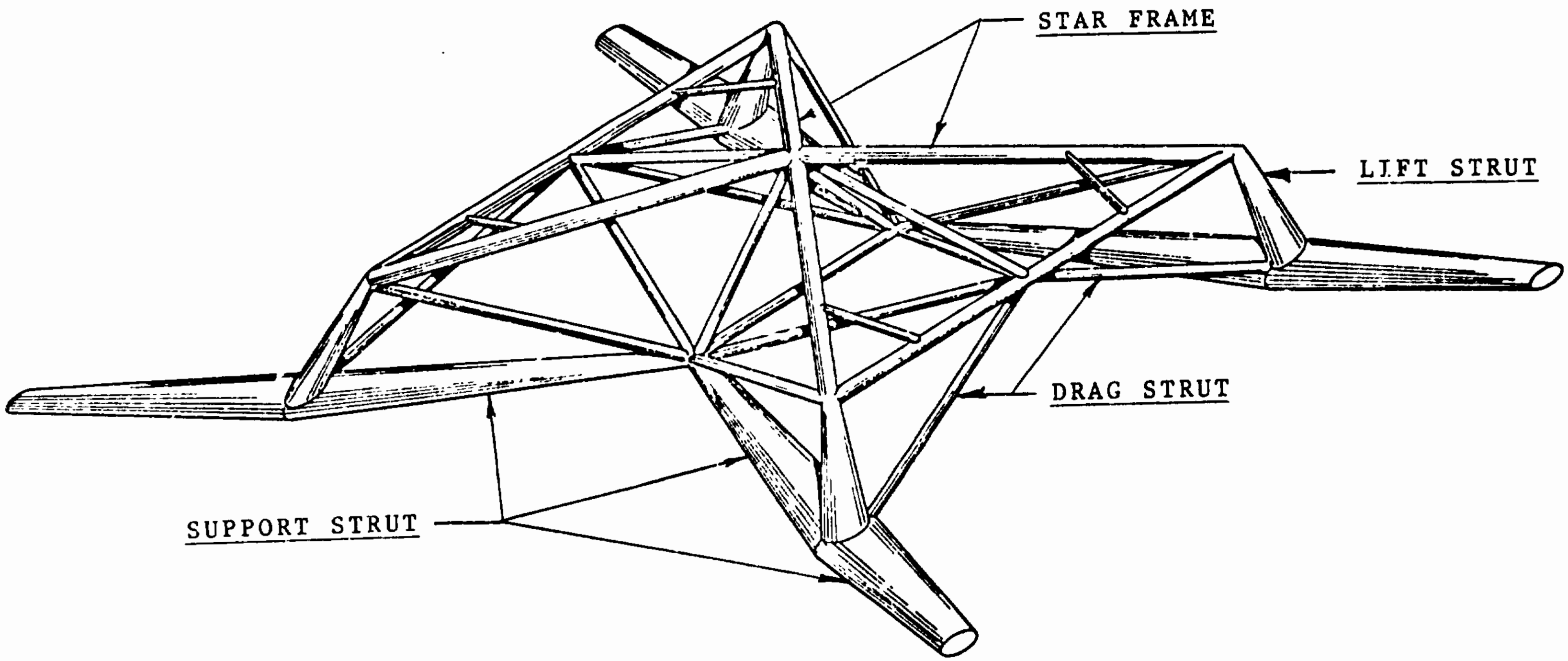
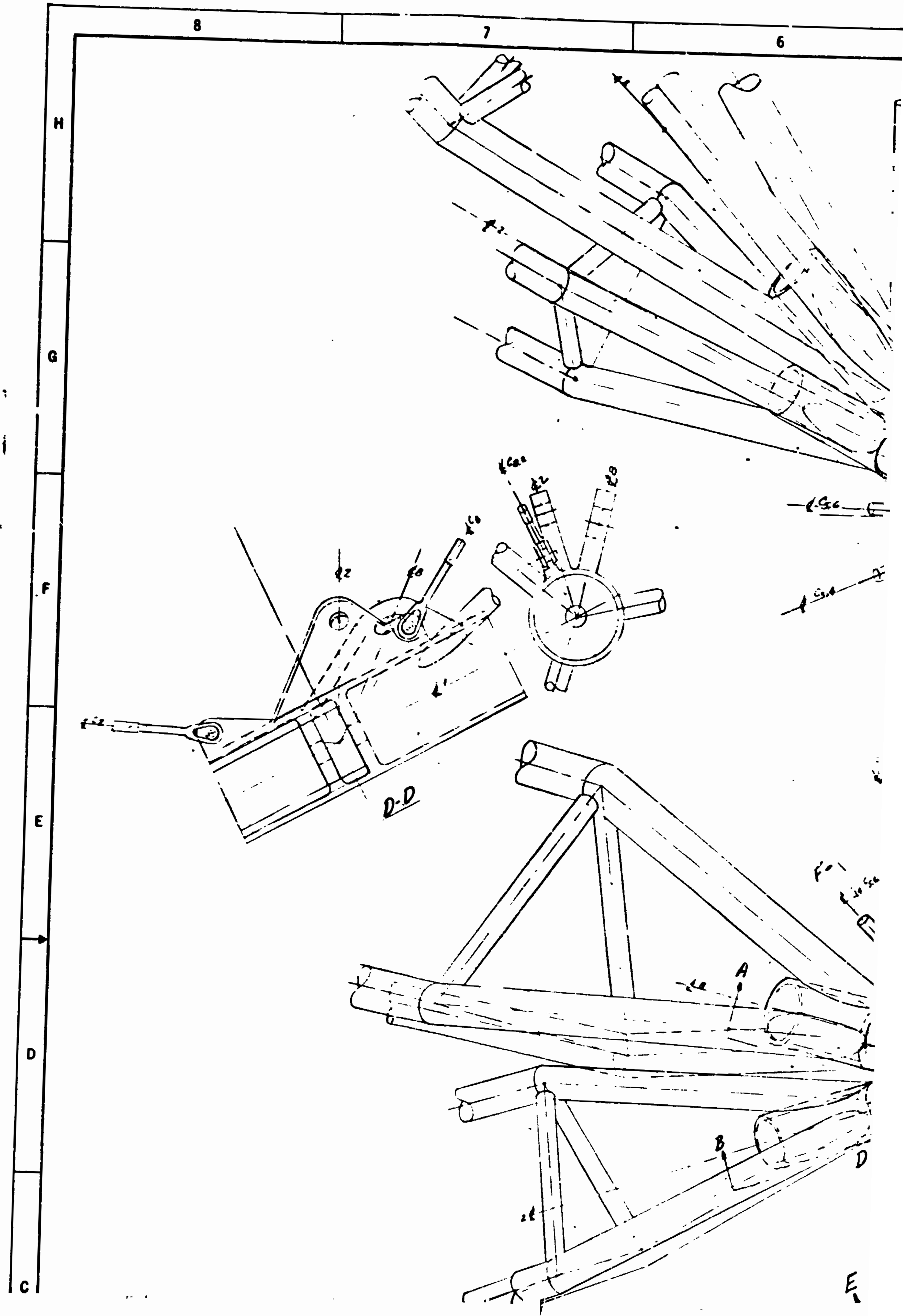
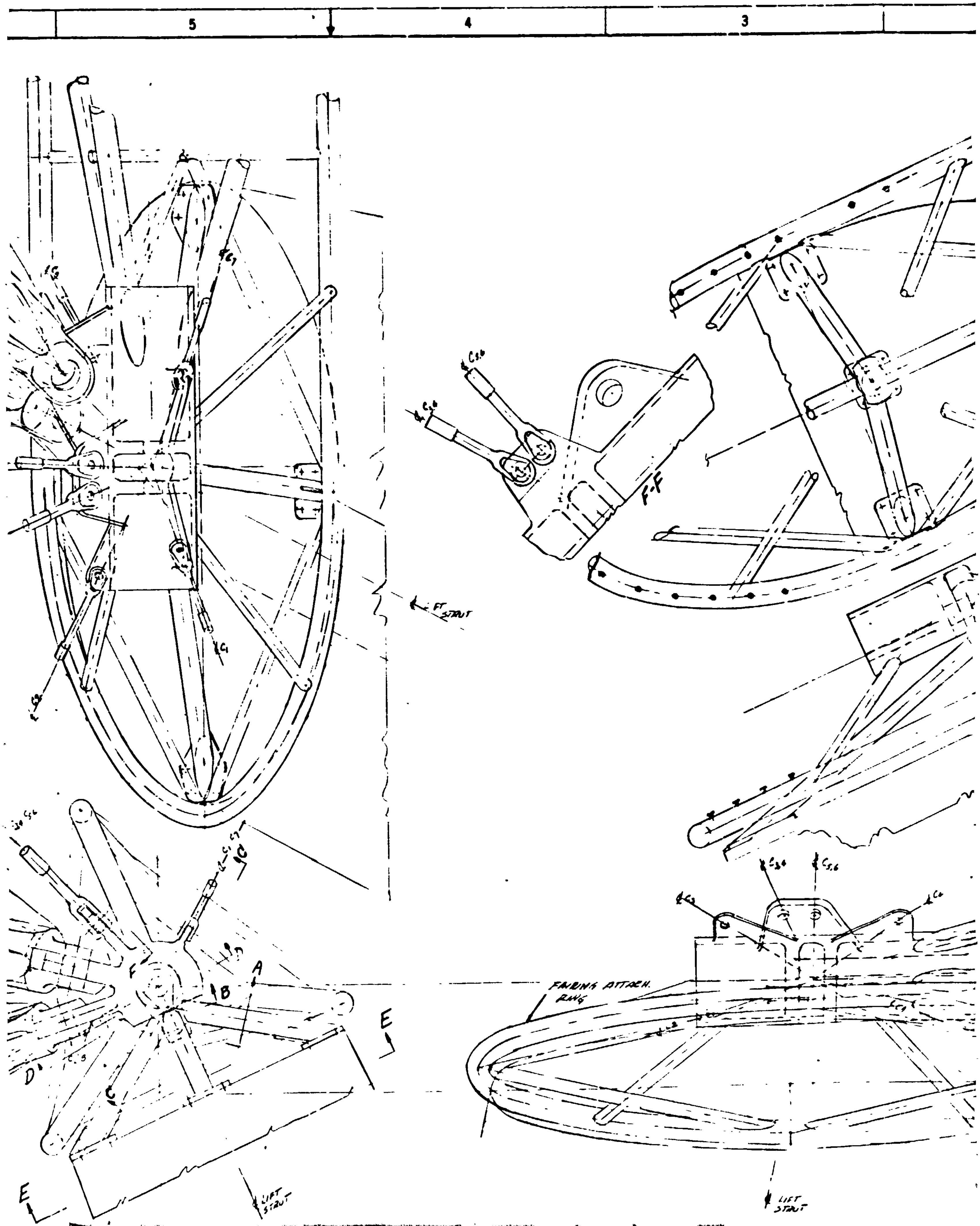


FIGURE 5.2 INTERCONNECTING STRUCTURE (CONSISTING OF FOUR LIFT STRUTS, FOUR DRAG STRUTS, FOUR SUPPORT STRUTS AND ONE INTERNAL STAR FRAME)

FIGURE 5.3 LF2 JOINT CONSTRUCTION
(GOODYEAR DRAWING NO. 76-304)

FOLDOUT FRAME 1



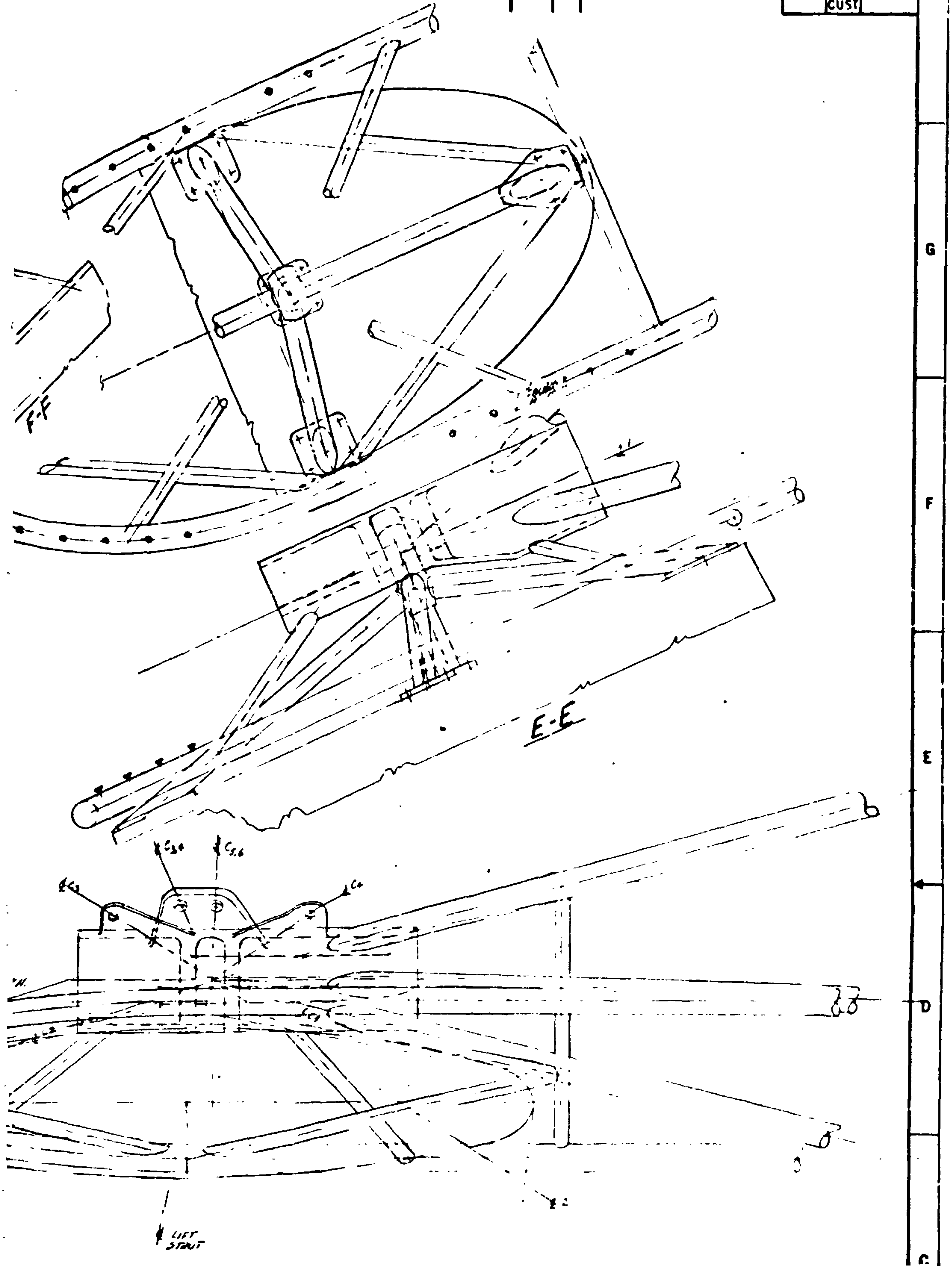

5
4
3
F-F
FT STRUT
A
B
C
D
E
E
F
FAIRING ATTACH. RING
LIFT STRUT
LIFT STRUT

FOLDOUT FRAME 3

3	2	1

REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED	
				BY	
				ENGR	
				CUST	



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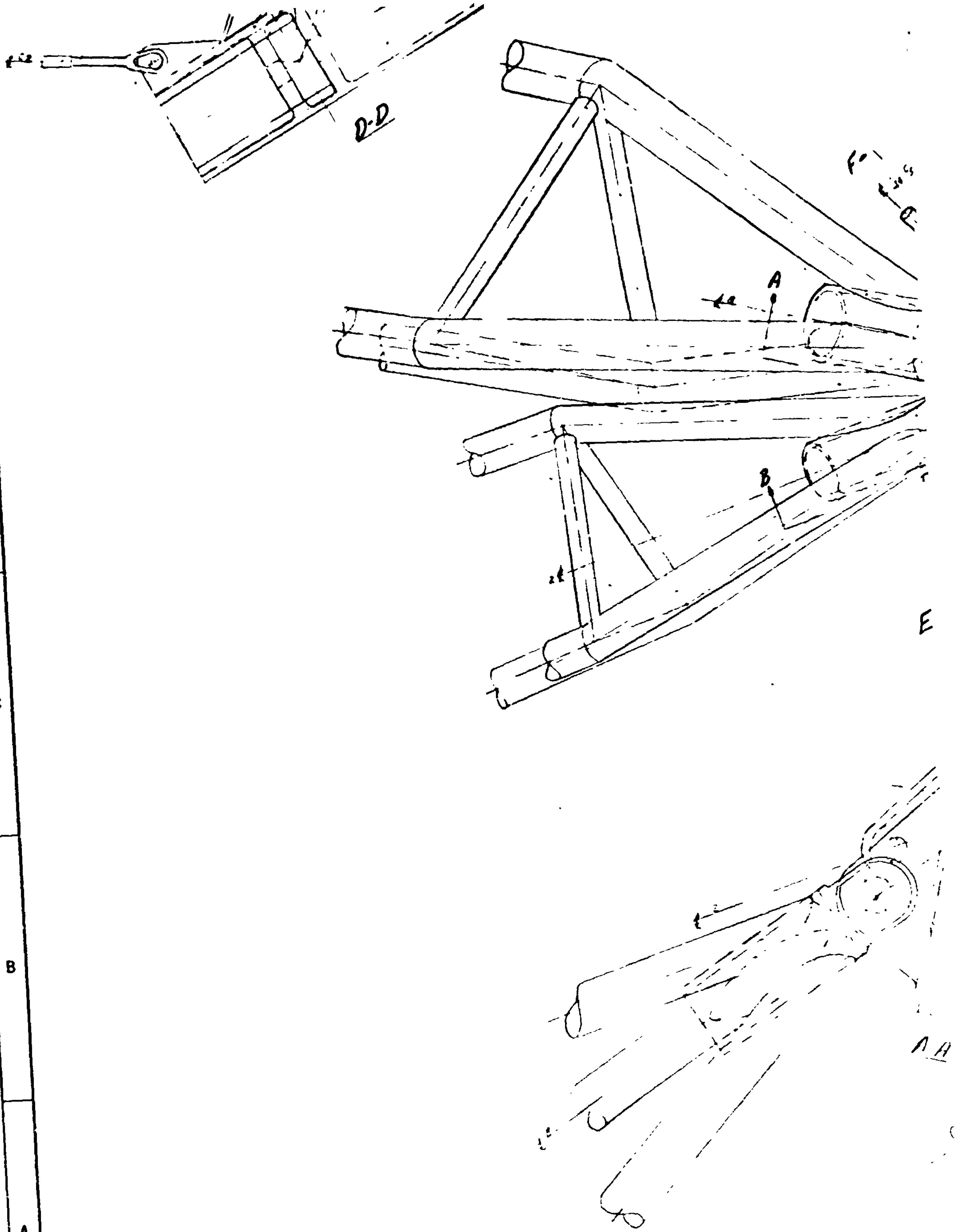
D

C

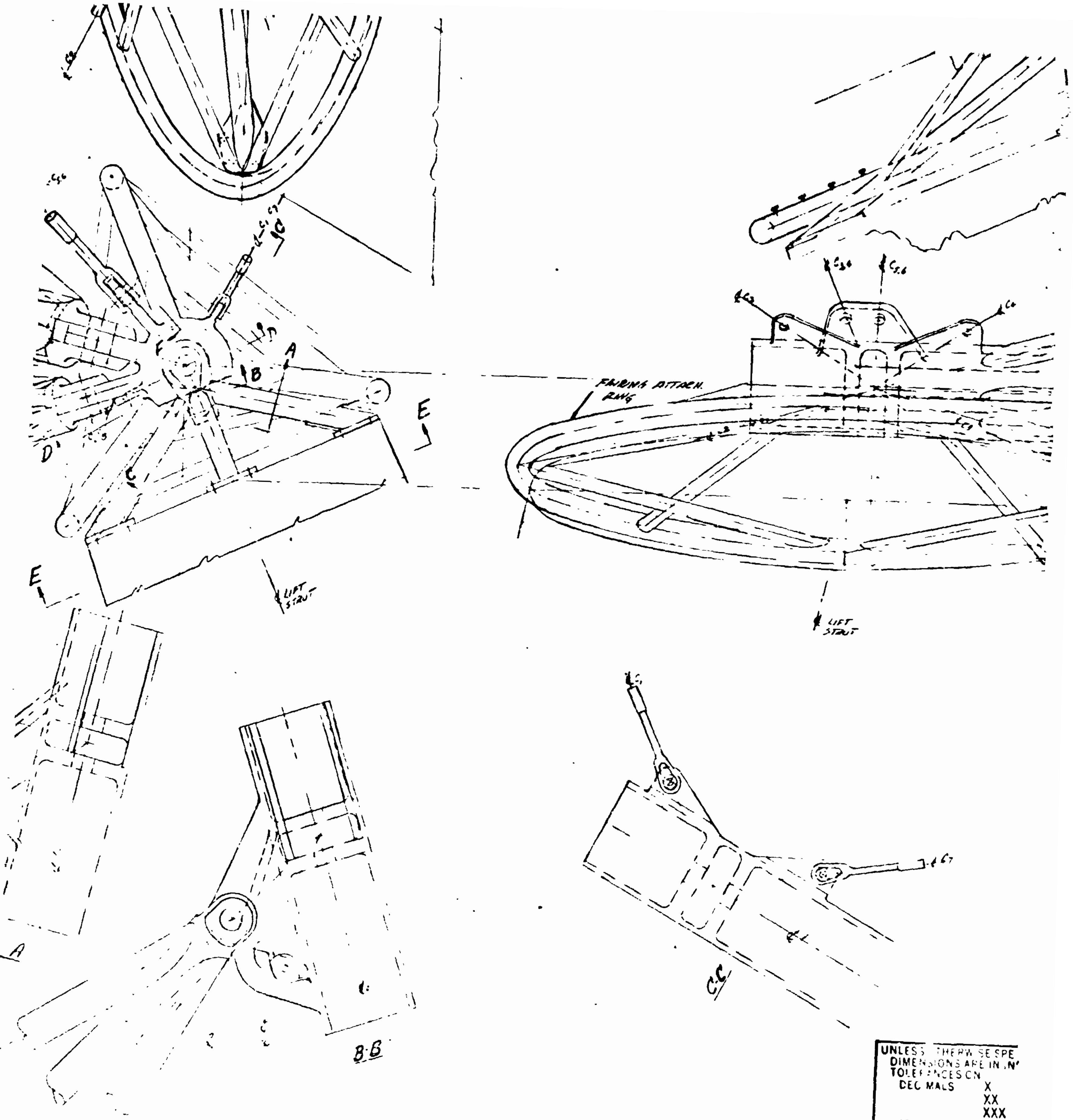
B

A



E-14-62 (2-66)

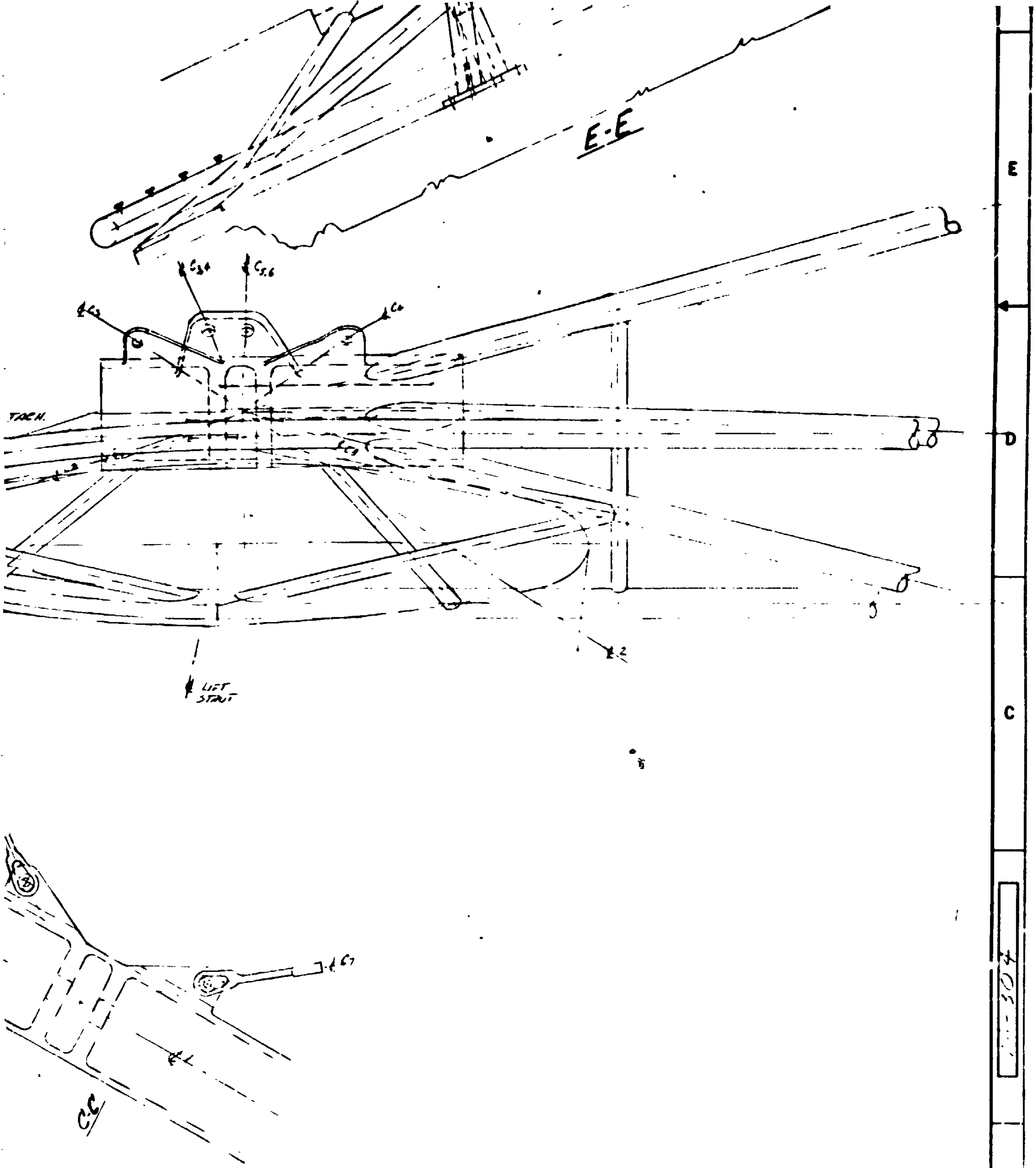
FOLDOUT FRAME 4



UNLESS OTHERWISE SPE...
DIMENSIONS ARE IN IN...
TOLERANCES ON
DECIMALS X
XX
XXX
ANGLES

MATERIAL

SEE ... FOR DR.
INTERPRETATION AND



UNLESS OTHERWISE SPECIFIED
DIMENSIONS ARE IN INCHES
TOLERANCES ON
DECIMALS X ± .1
XX ± .04
XXX ± .010
ANGLES ± 1°

MATERIAL

SEE G.. 9-60 FOR DRAWING INTERPRETATION STANDARDS

CONTR		
WTS		
STRESS		
ENGR		
CHECK		
DRAWN		

GAC APPROVAL

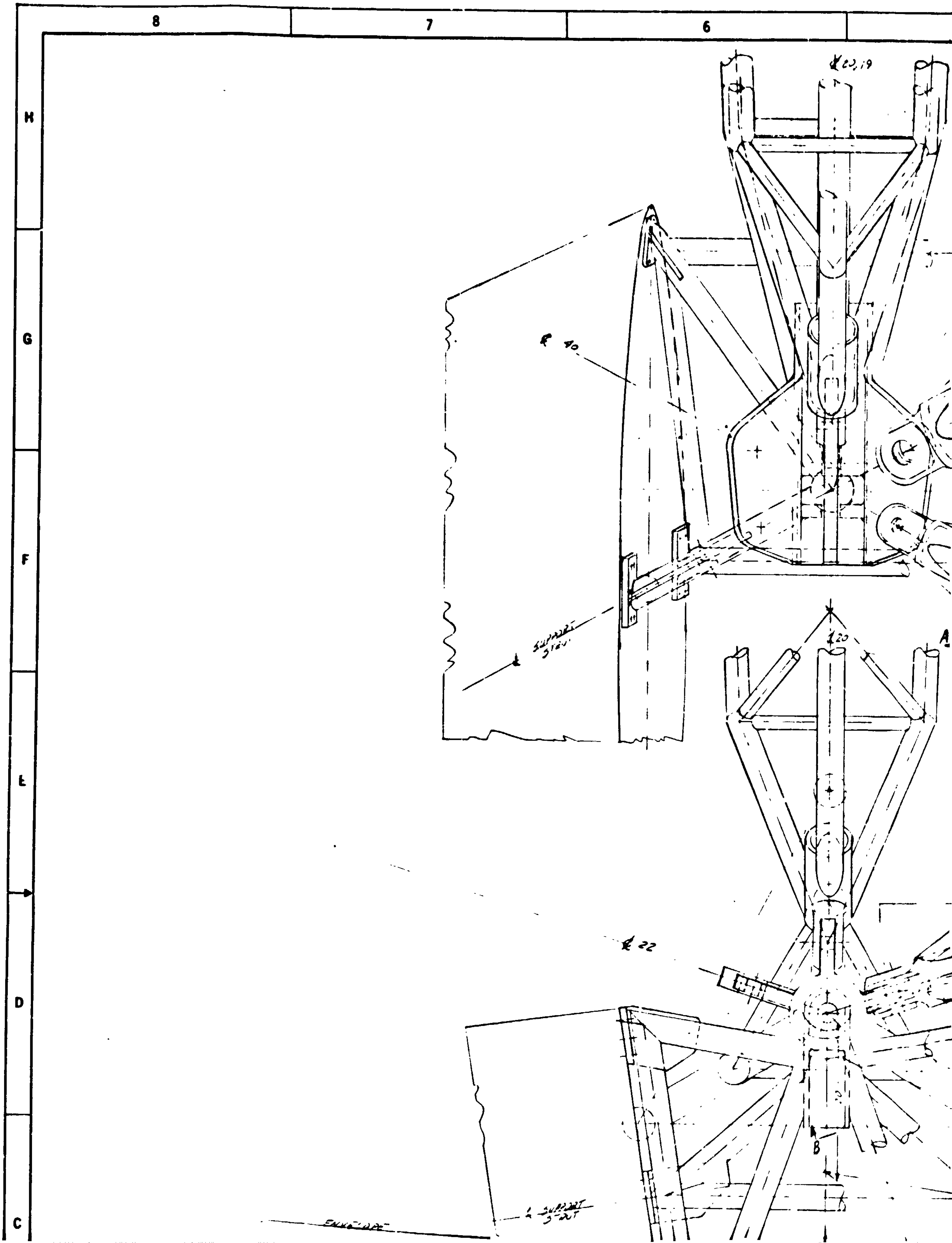
CUSTOMER APPROVAL

GOODYEAR AEROSPACE CORP
AKRON OHIO

SIZE	CODE IDENT NO	DRAWING NO
E	25500	

SCALE | SHEET 1 OF

FIGURE 5.4 F-1, A-1 JOINT CONSTRUCTION
(GOODYEAR DRAWING NO. 76-305)



8
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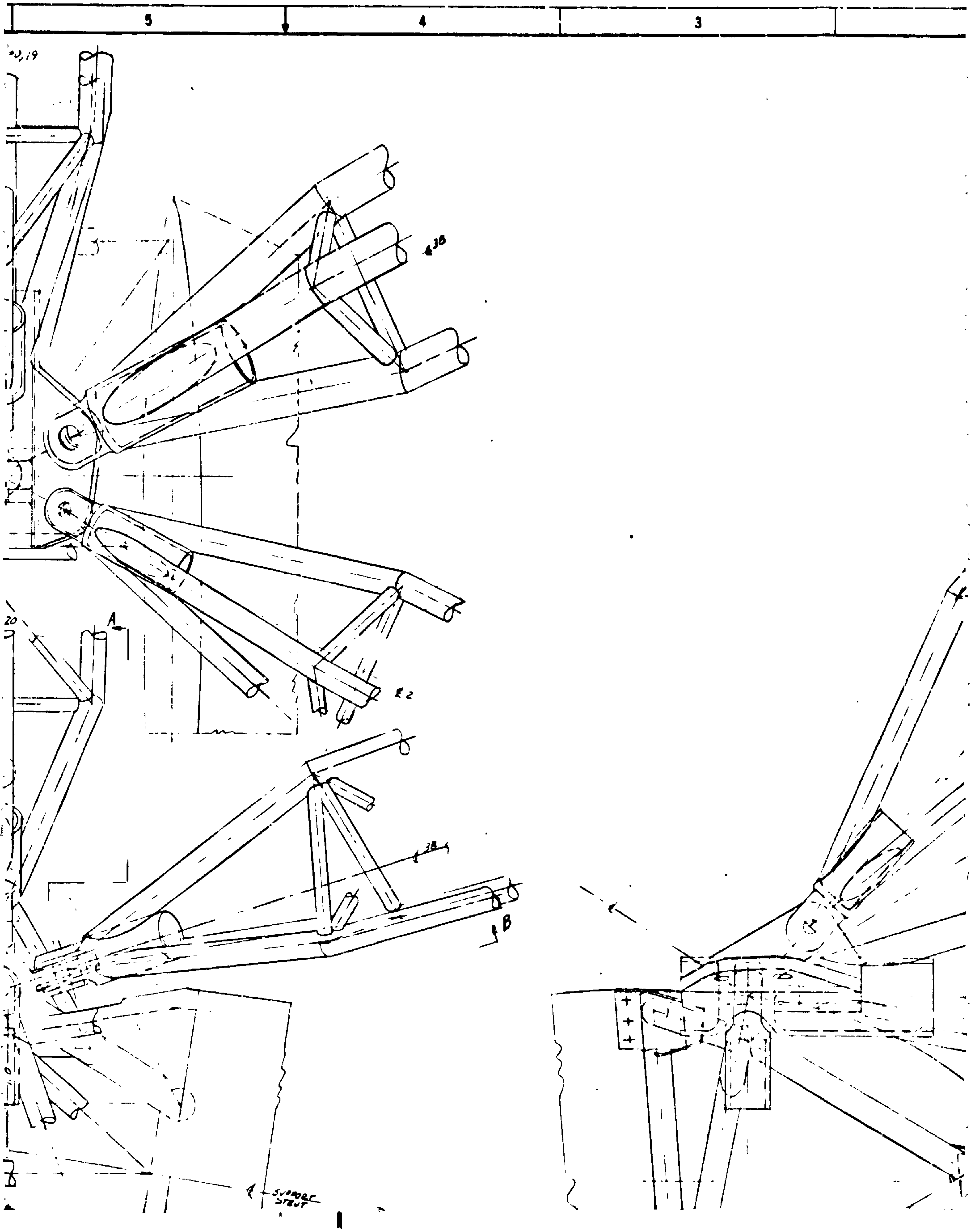
C
℄ 20,19
℄ 40
℄ SUPPORT STRUT
℄ 20
A
℄ 22
B
℄ SUPPORT STRUT

FOLDOUT FRAME



3	2	1

REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED	
				BY	
				ENGR	
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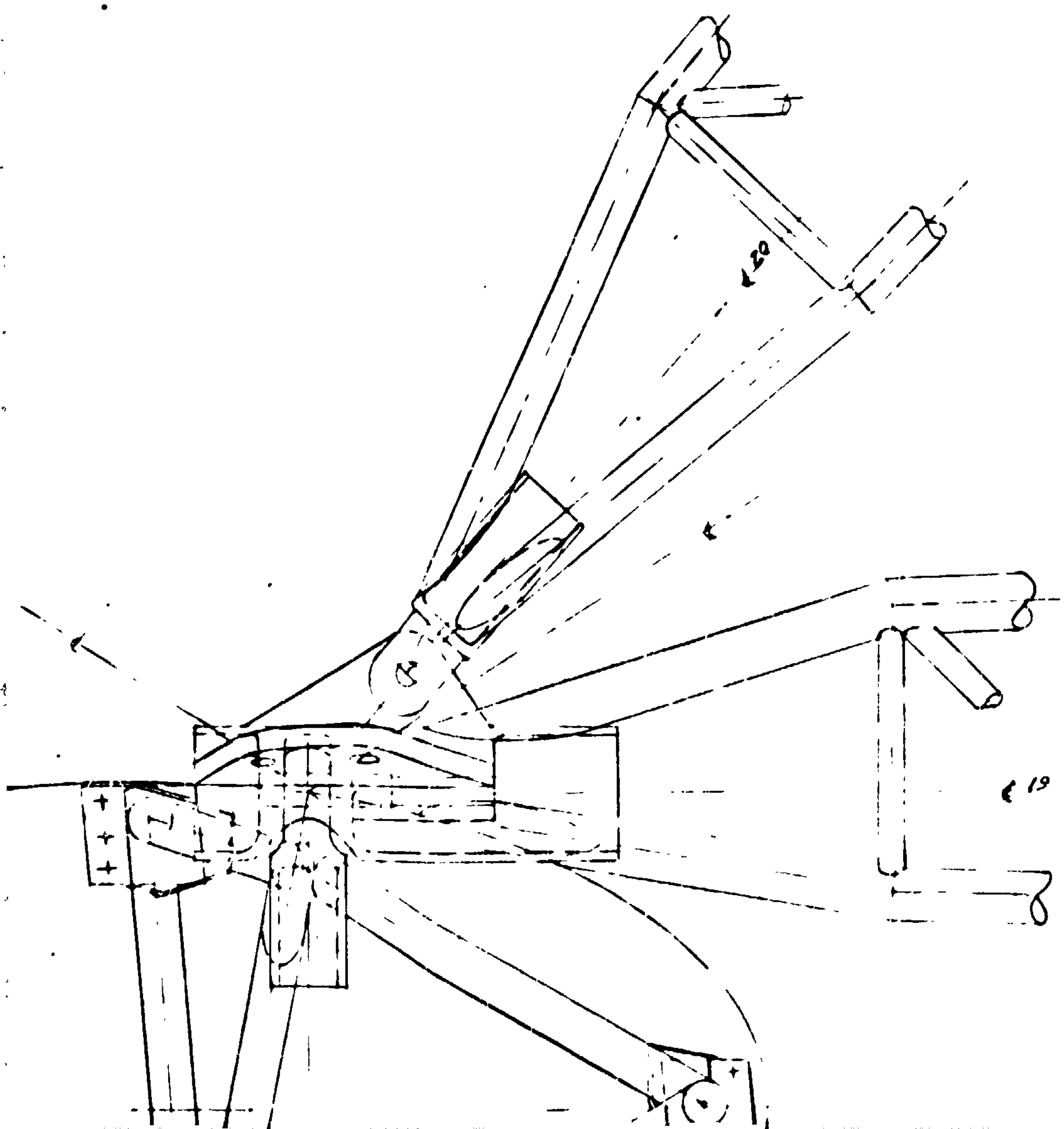

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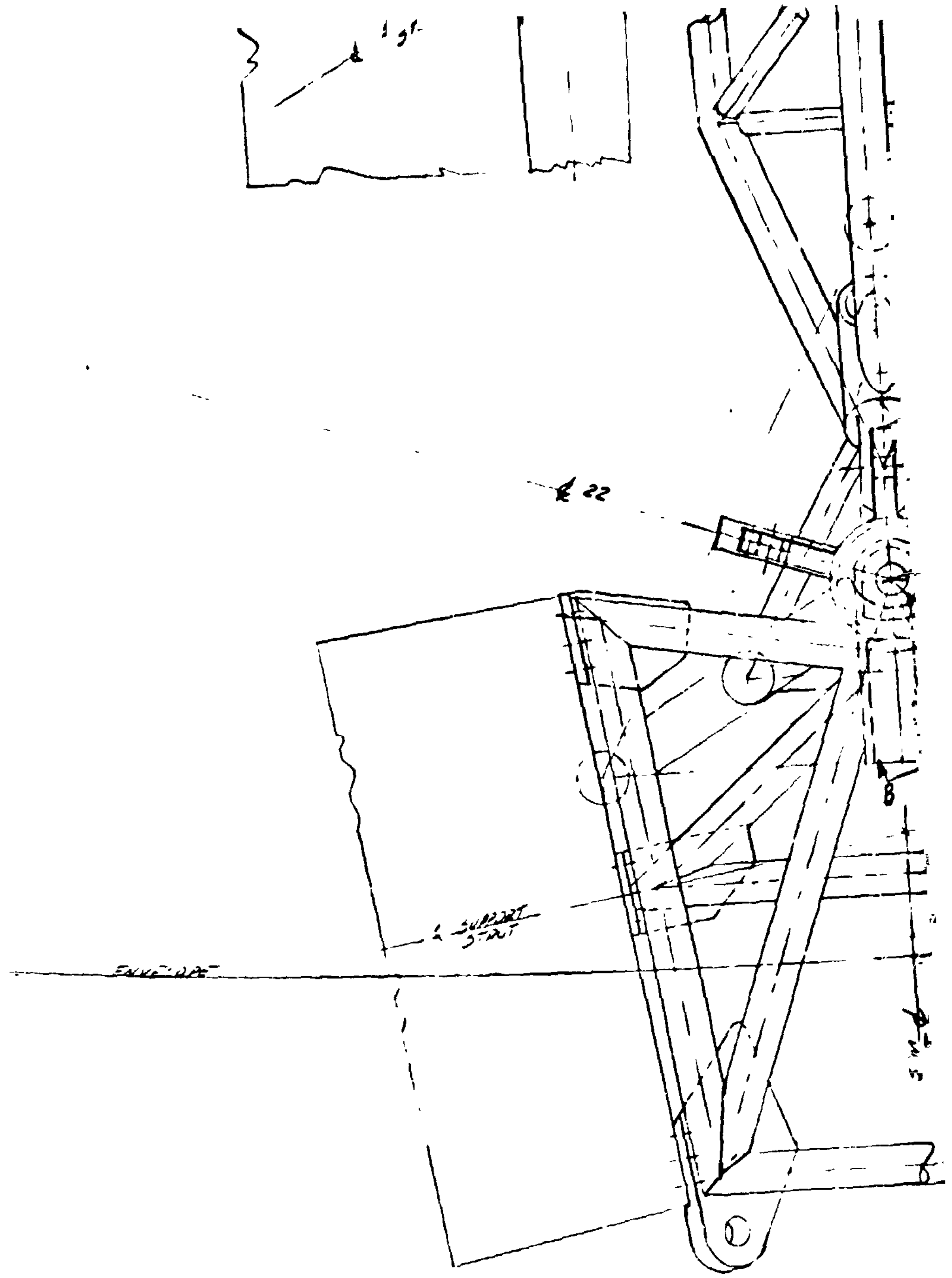

E-IN-62 (2-66)

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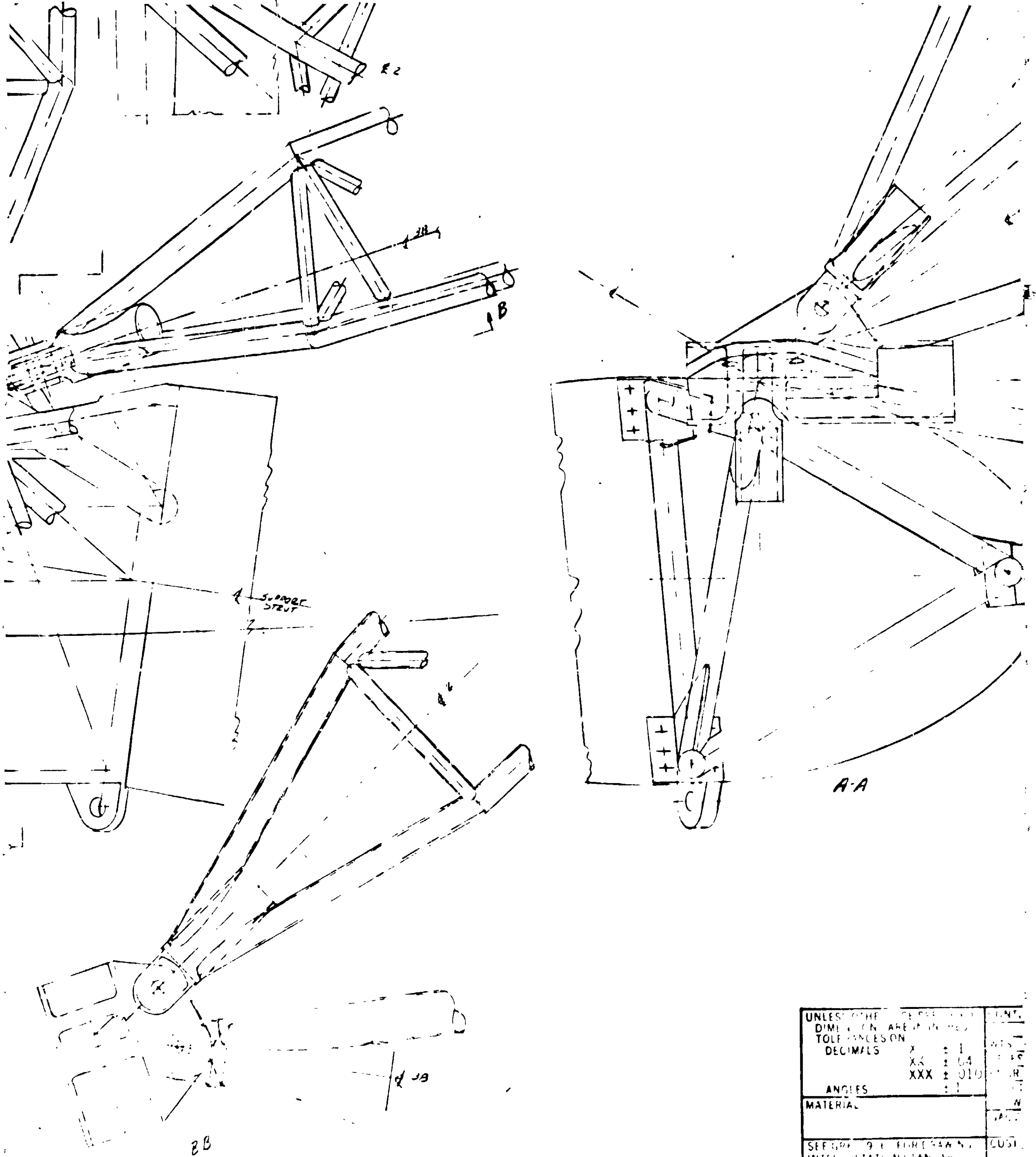
OF POOR QUALITY

FOLDOUT FRAME 4



UNLESS OTHE[illegible] DIMEN[illegible] ARE IN [illegible] TOLERANCES ON		
DECIMALS	X	± .1
	XX	± .04
	XXX	± .010
ANGLES		± 1
MATERIAL		
SEE GR[illegible] 9 [illegible] FOR DRAWING INTERPRETATION STANDARDS		

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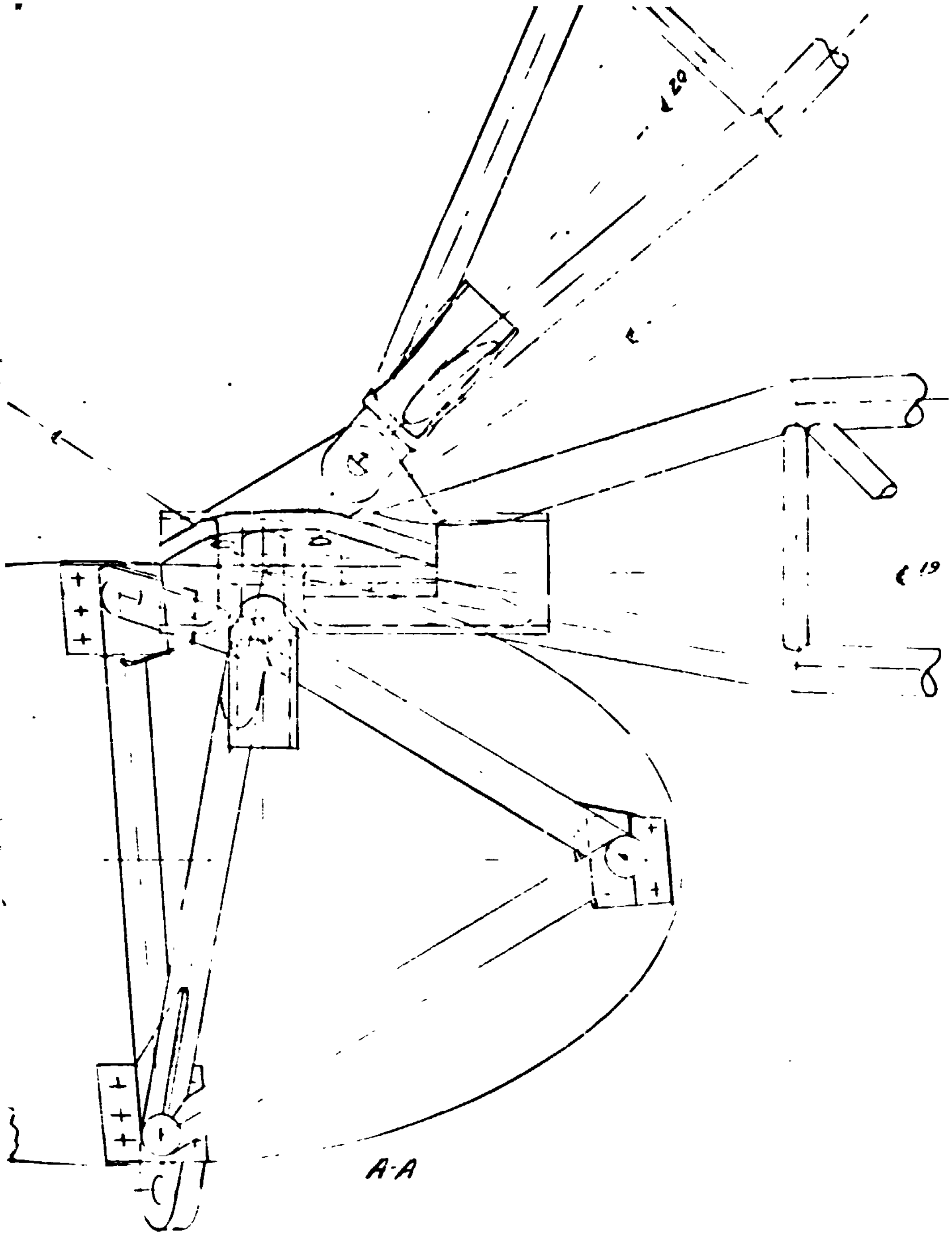

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16-305

UNLESS OTHERWISE SPECIFIED
DIMENSIONS ARE IN INCHES
TOLERANCES ON
DECIMALS X ± .1
XX ± .04
XXX ± .010
ANGLES ± 1°

MATERIAL

SEE GPS 1-9-60 FOR DRAWING
INTERPRETATION STANDARDS

CONTR

GAC APPROVAL

CUSTOMER APPROVAL

GOODYEAR AEROSPACE CORP
AKRON OHIO

SIZE E
CODE IDENT NO 25500

DRAWING NO

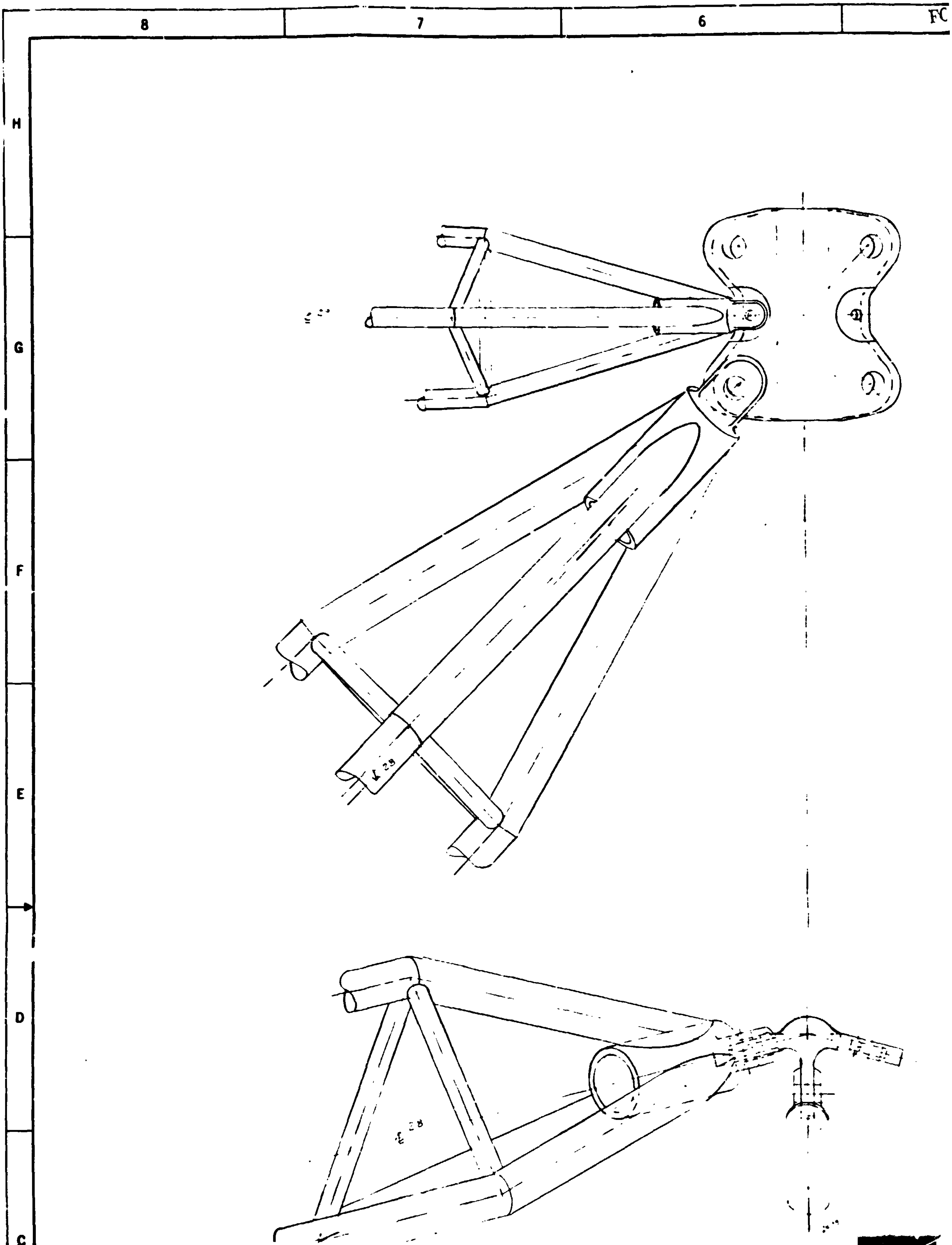
SCALE

SHEET 1 OF

FOLDOUT FRAME 2

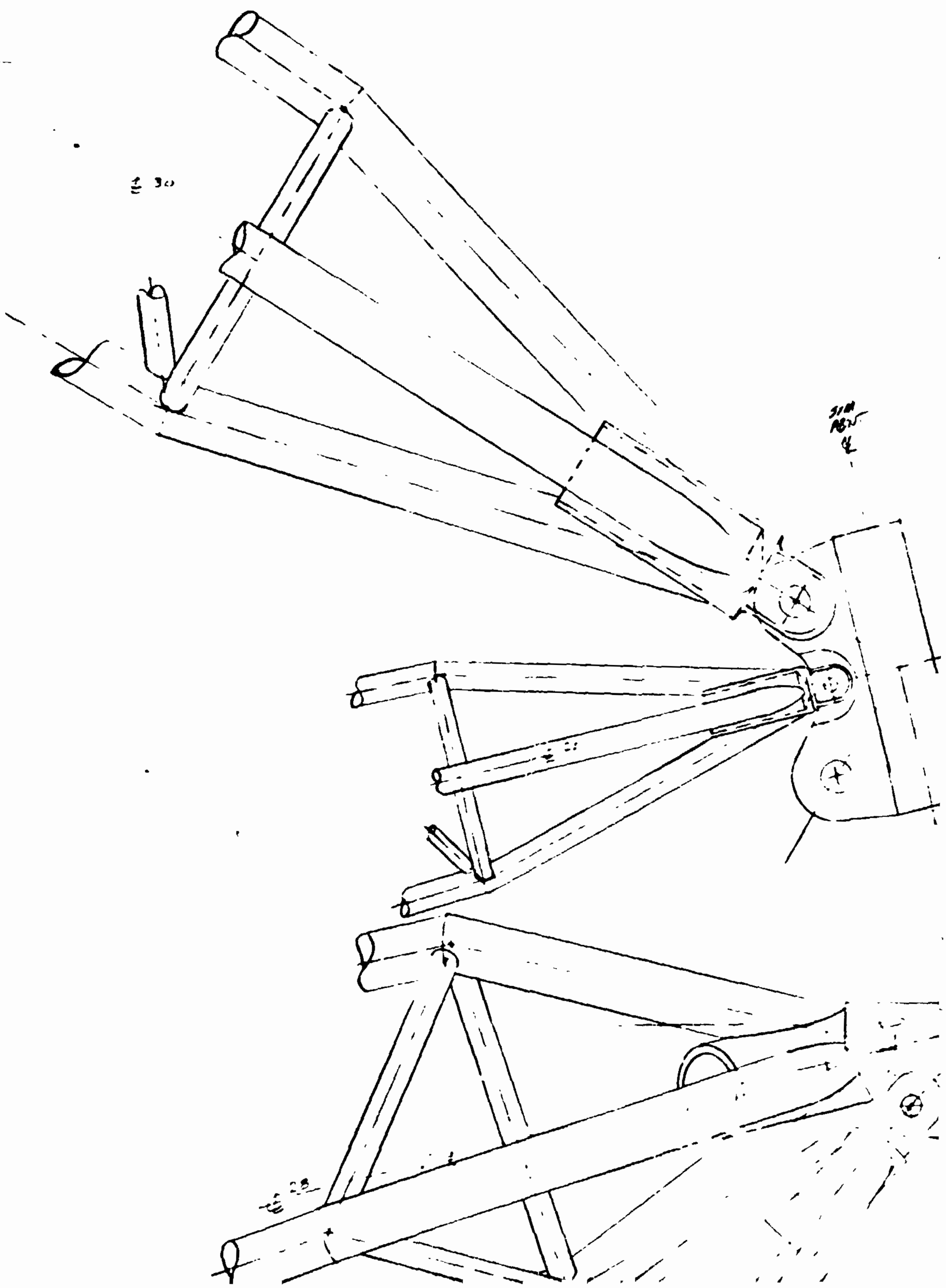
FIGURE 5.5 C2 JOINT CONSTRUCTION
(GOODYEAR DRAWING NO. 76-323)

FOLDOUT FRAME



FOLDOUT FRAME 2

4	3	2



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REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED	
				BY	
				ENGR	
				CUST	

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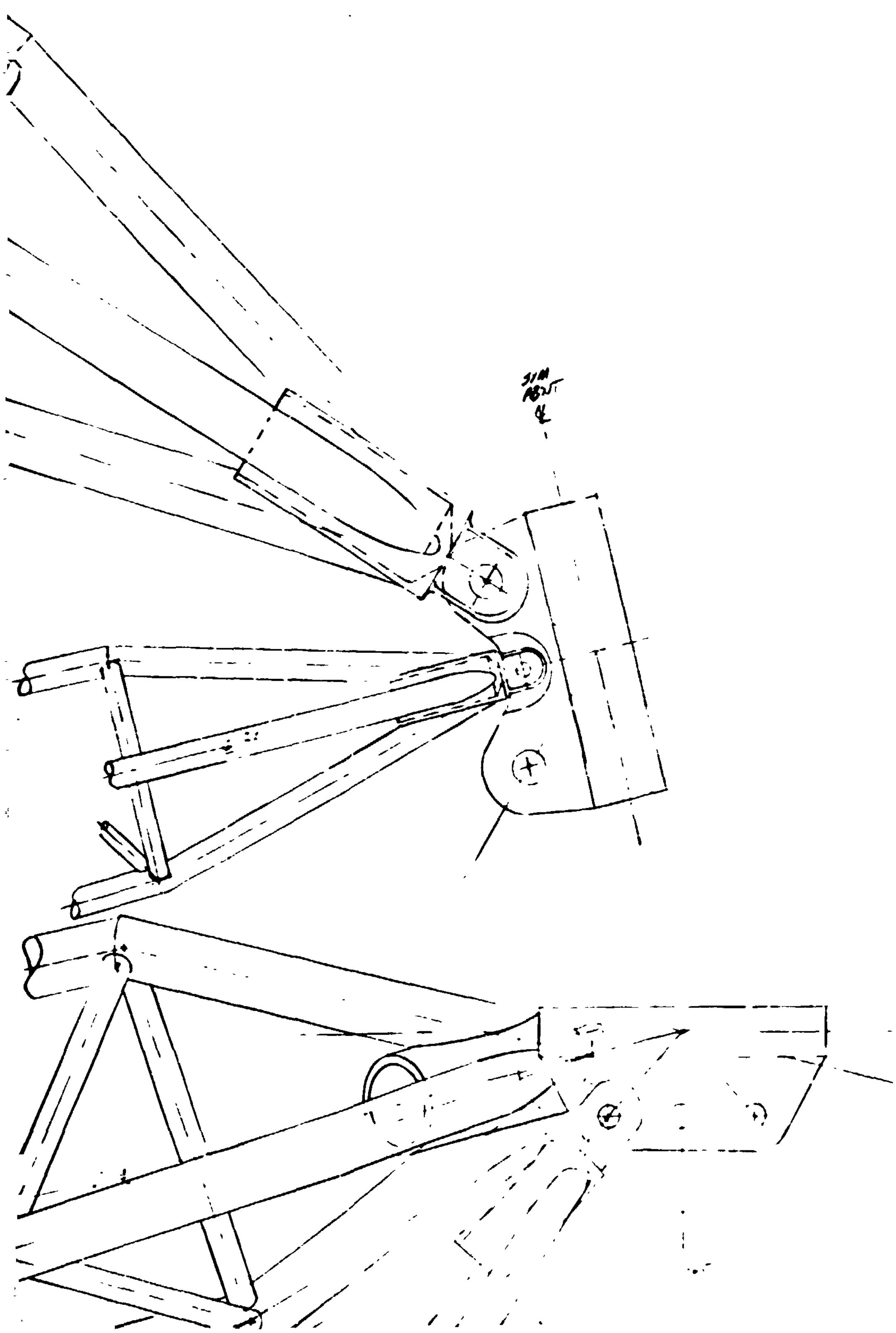

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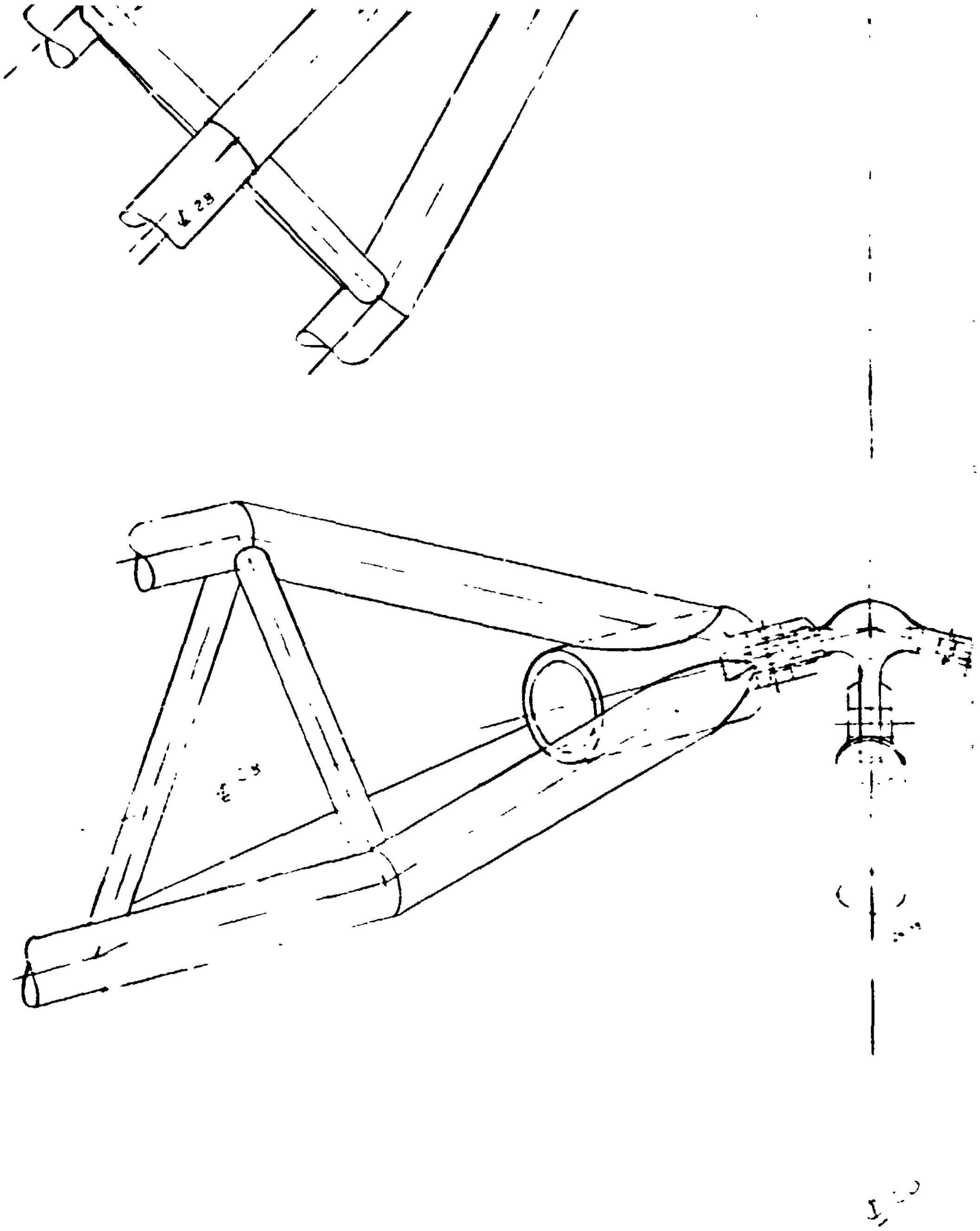
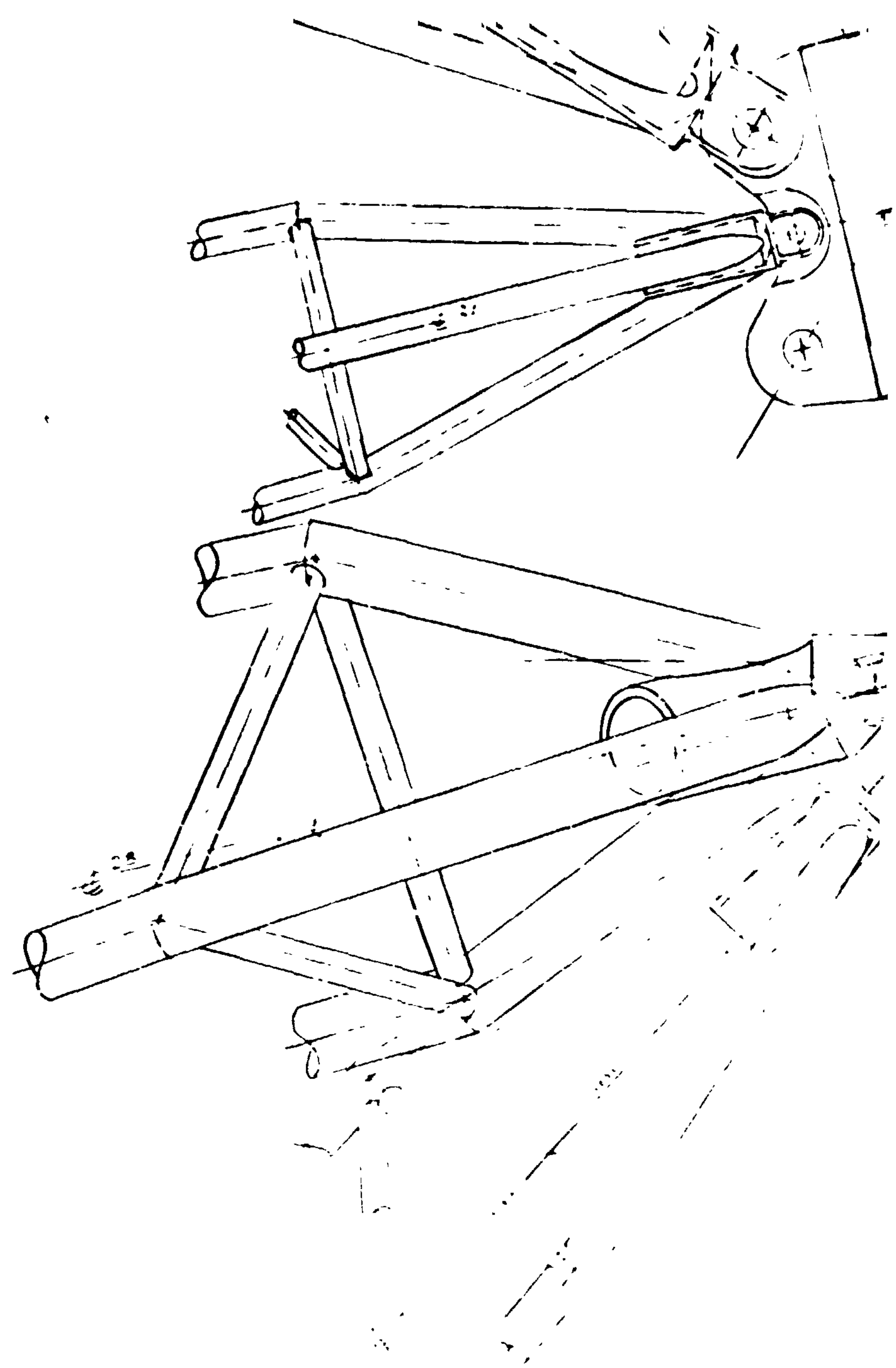


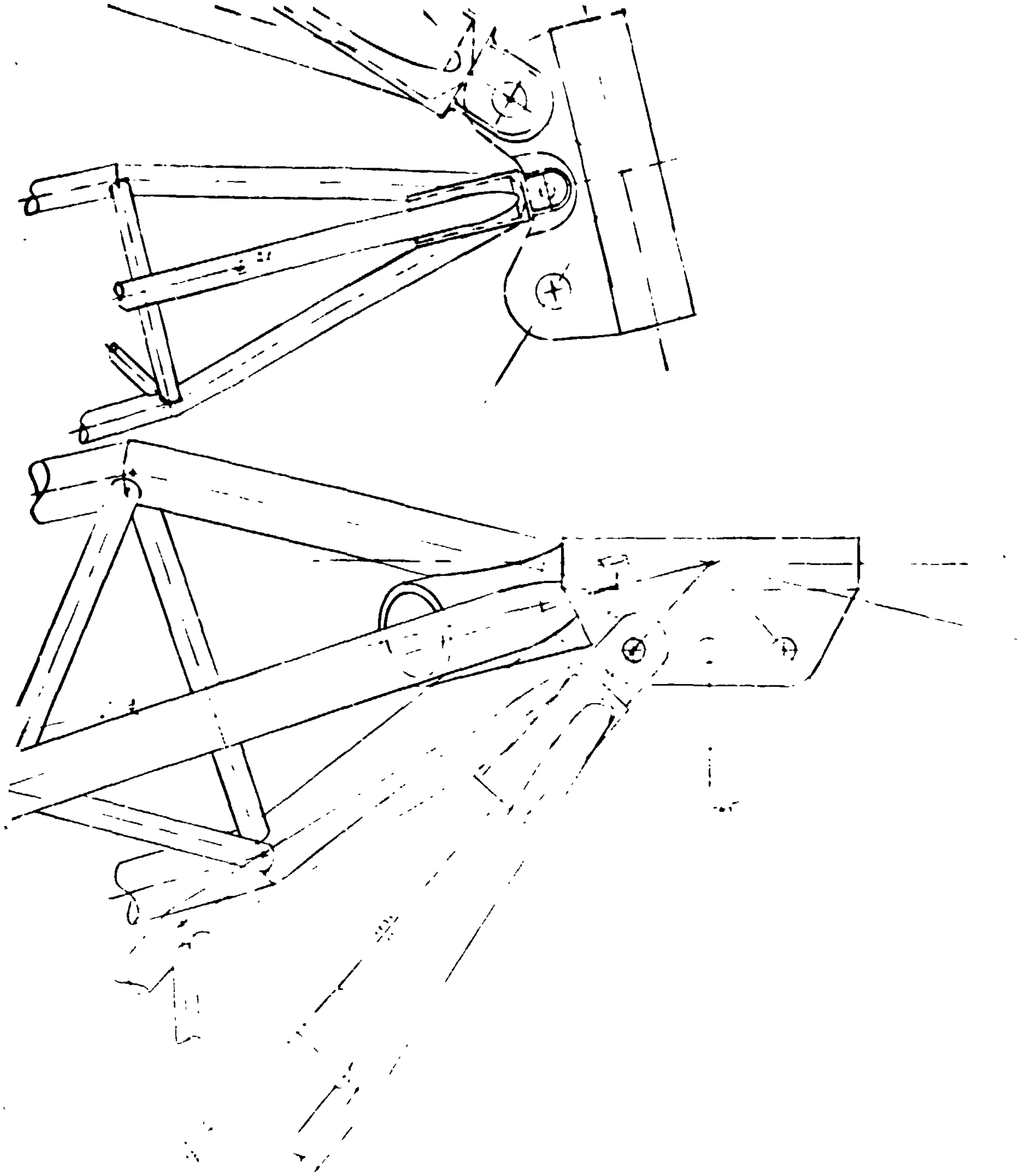
E-IN-62 (2-66)

FOLDOUT FRAME 4



UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES TOLERANCES ON DECIMALS X ± 1 XX ± 04 XXX ± 010 ANGLES ± 1°	CONTR
MATERIAL	GAC AP.
SEE GR50-9-61 FOR DRAWING INTERPRETATION STANDARDS	CUSTOM

FRAME 5



UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES TOLERANCES ON DECIMALS X ±.1 XX ±.04 XXX ±.010 ANGLES ±1°	CONTR	GOODYEAR AEROSPACE CORP AKRON OHIO
	WTS	
	STRESS	
	ENGR	
	CHECK	
MATERIAL	DRAWN	
	GAC APPROVAL	SIZE E · CODE IDENT NO 25500 · DRAWING NO
SEE GR50-9-6[illegible] FOR DRAWING INTERPRETATION STANDARDS	CUSTOMER APPROVAL	SCALE · SHEET 1 OF

E

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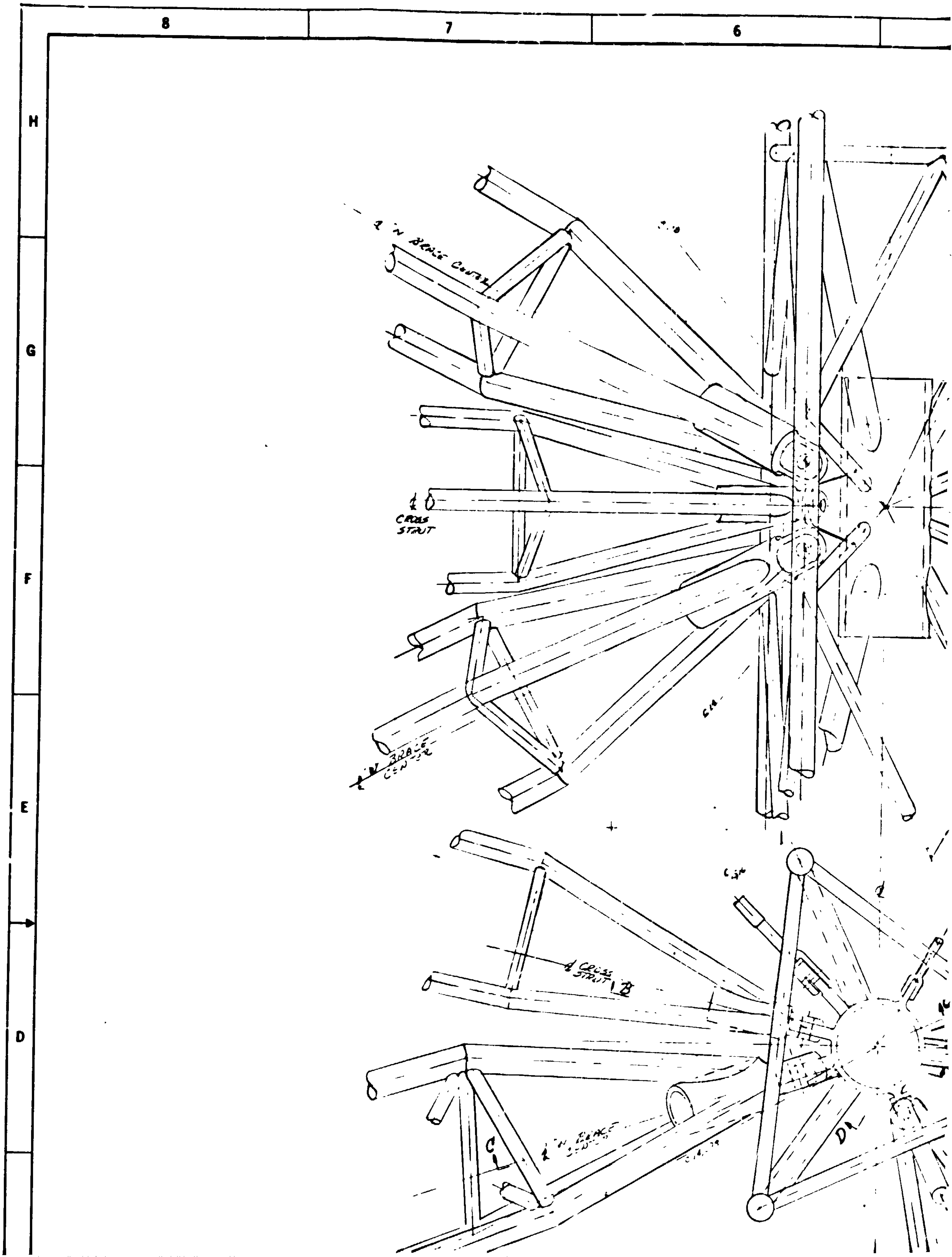
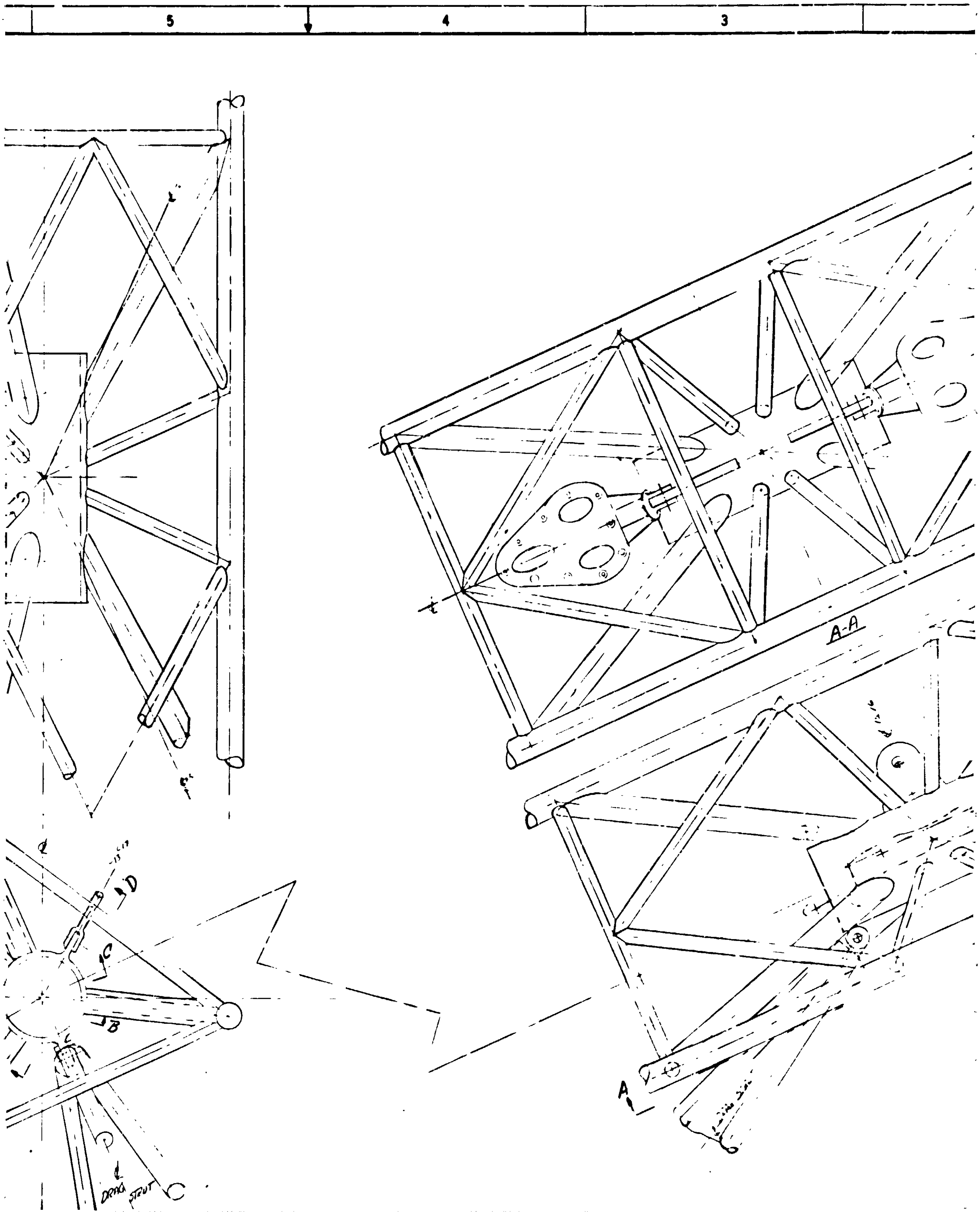
6

FIGURE 5.6 L2 JOINT CONSTRUCTION
(GOODYEAR DRAWING NO. 76-330)

FOLDOUT FRAME



FOLDOUT FRAME 2

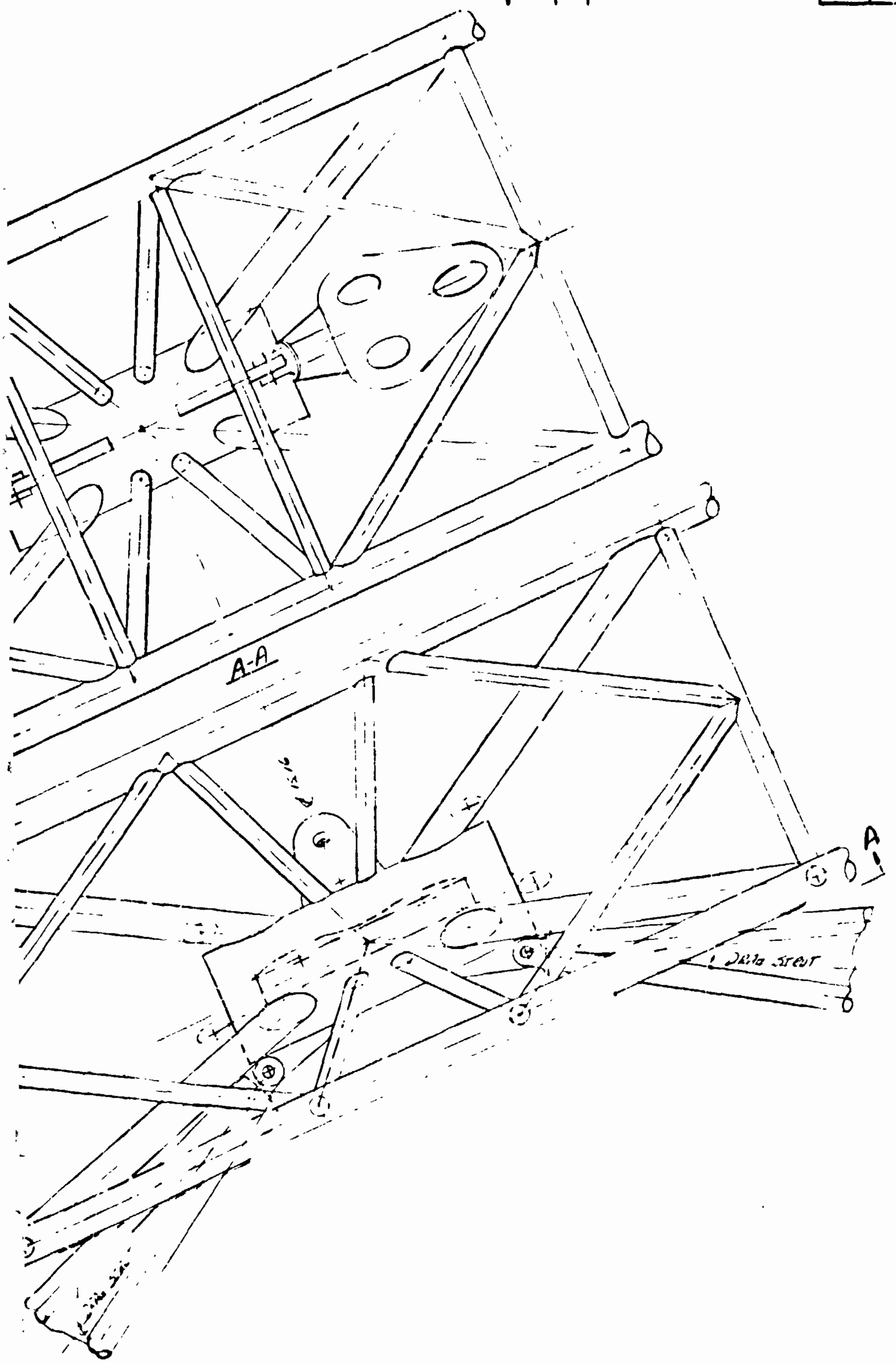


FOLDOUT FRAME

3	2	1

REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED	
				BY	
				ENGR	
				CUST	



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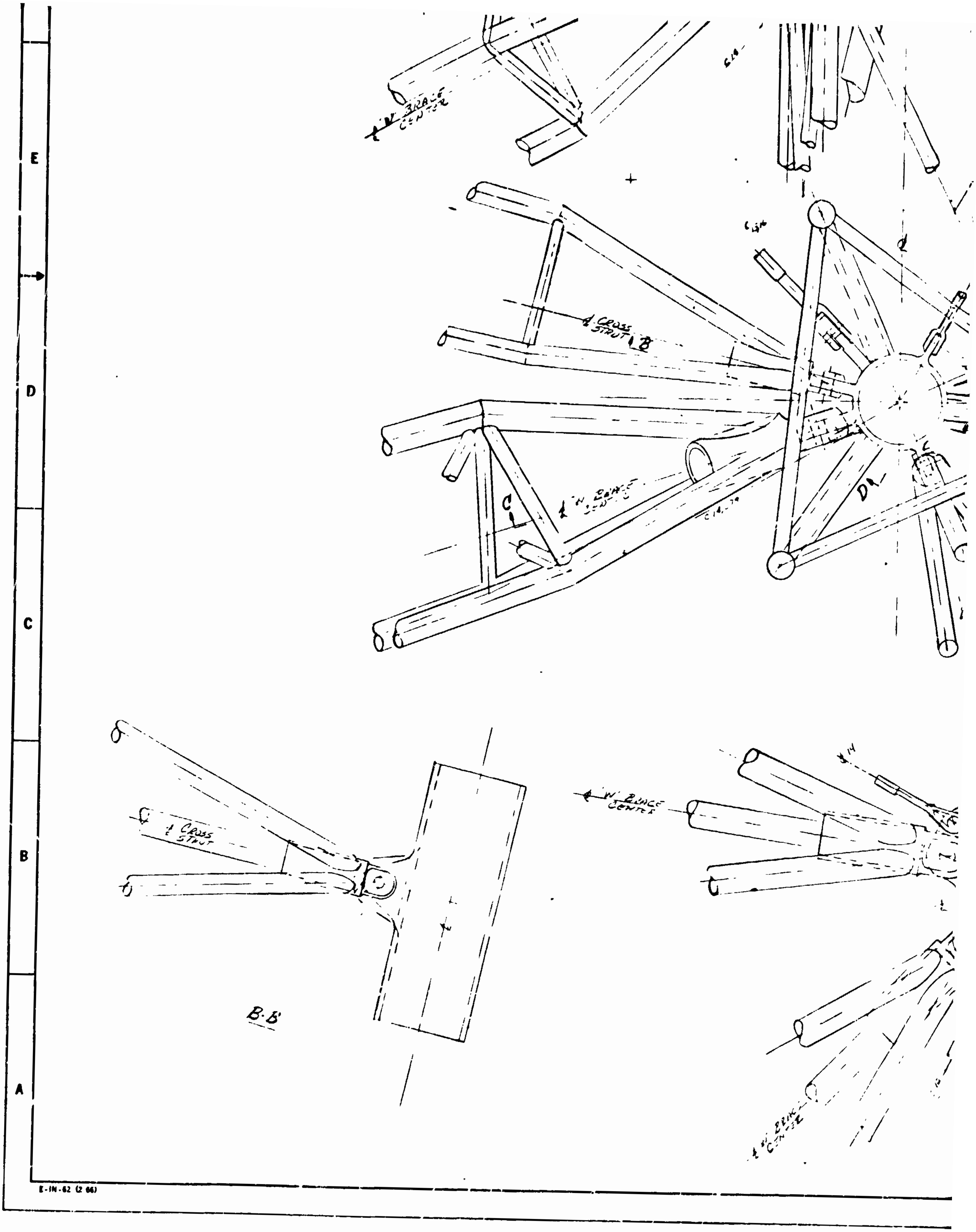
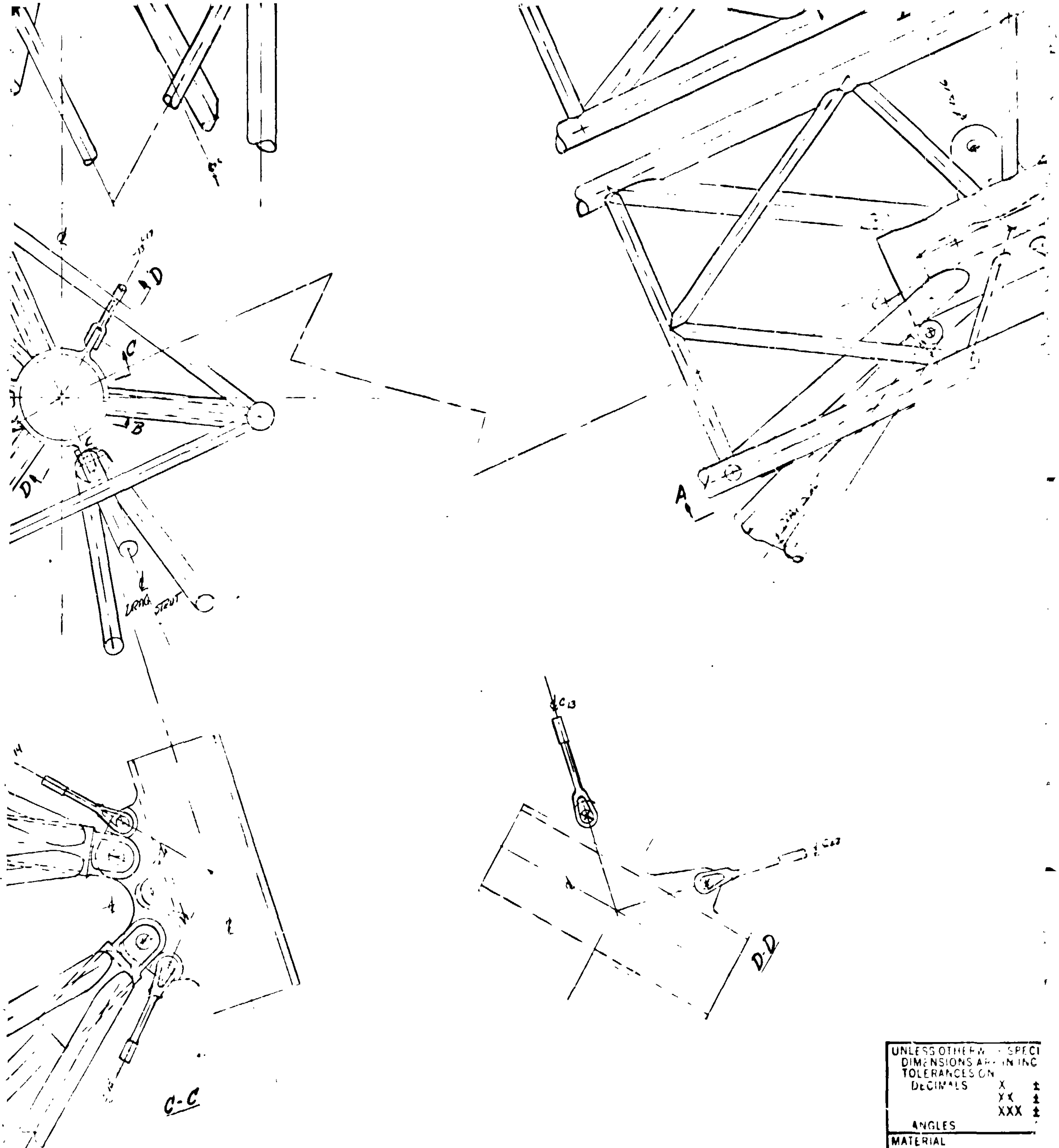


FOLDOUT FRAME 4

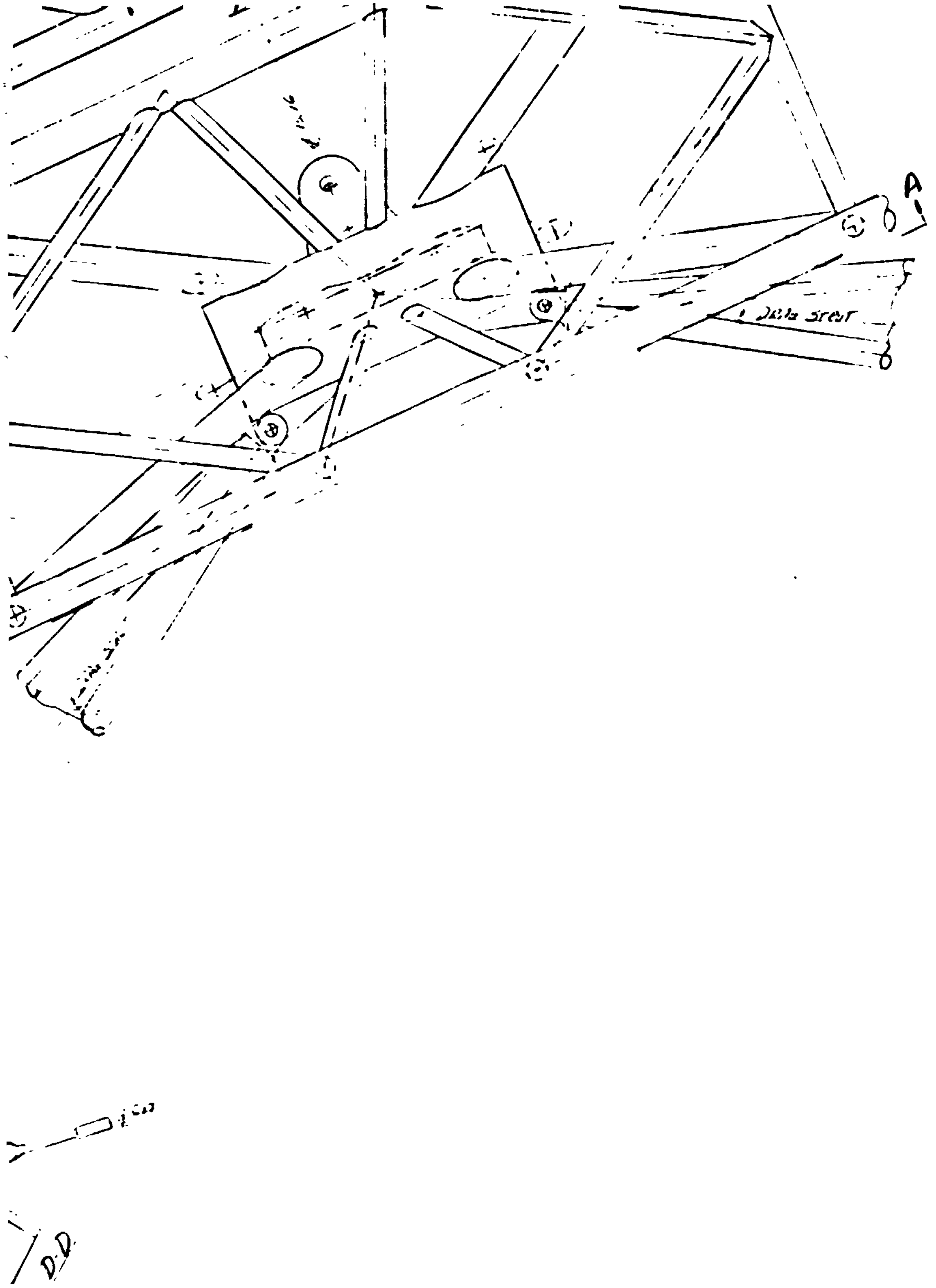
FOLDOUT



UNLESS OTHERW... SPECI...
DIMENSIONS AR... IN INC...
TOLERANCES ON
DECIMALS X ±
XX ±
XXX ±
ANGLES
MATERIAL

SEE GR60-9 ... INTERPRETATION STANDA...

FOLDOUT FRAME 5



UNLESS OTHERWISE SPEC'FIED DIMENSIONS ARE IN INCHES TOLERANCES ON DECIMALS X ± .1 .XX ± .04 .XXX ± .010 ANGLES ± 1°	CONTR / WTS / STRESS / ENGR / CHECK / DRAWN	GOODYEAR AEROSPACE CORP AKRON OHIO	
MATERIAL	GAC APPROVAL	E	DRAWING NO
SEE GR60-9-60 FOR DRAWING INTERPRETATION STANDARDS	CUSTOMER APPROVAL	SCALE	SHEET 1 OF

FOLDOUT FRAME 6

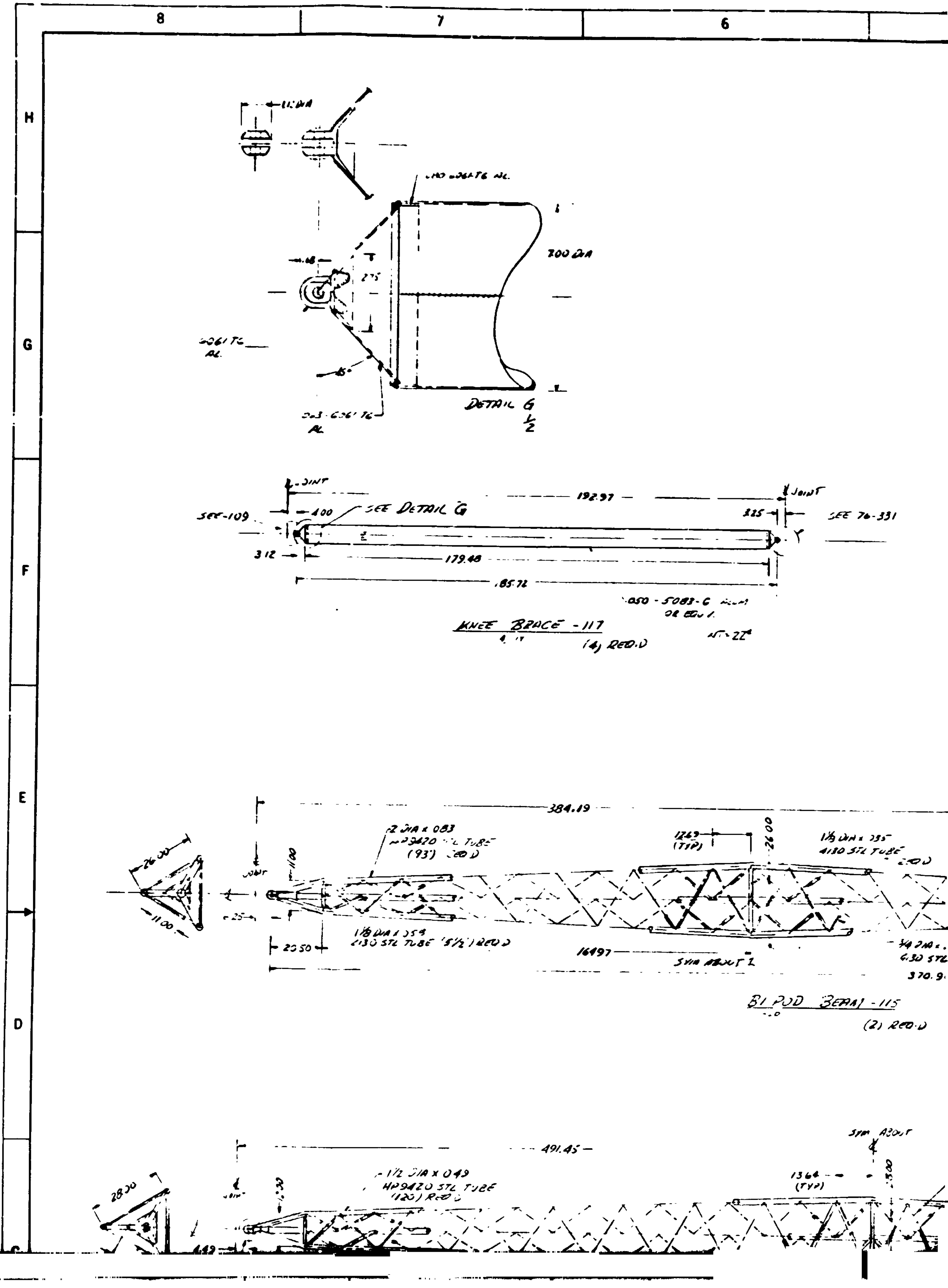
FIGURE 5.7 GIRDER ASSEMBLY
(GOODYEAR DRAWING NO. 76-332)

NOTE: 1.0 ft = 3.048×10^{-1} m

1.0 in = 2.54×10^{-2} m

1.0 lb = 4.536×10^{-1} kg

FOLDOUT FRAME 1



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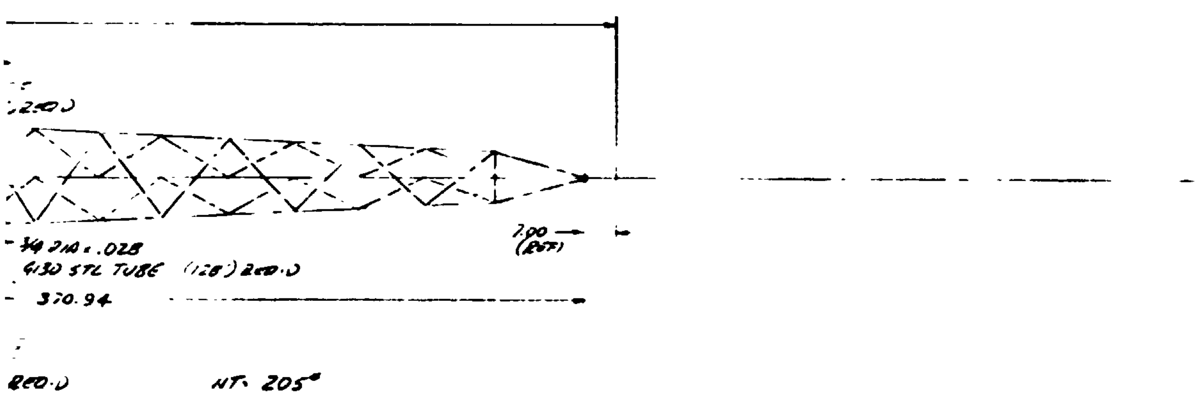
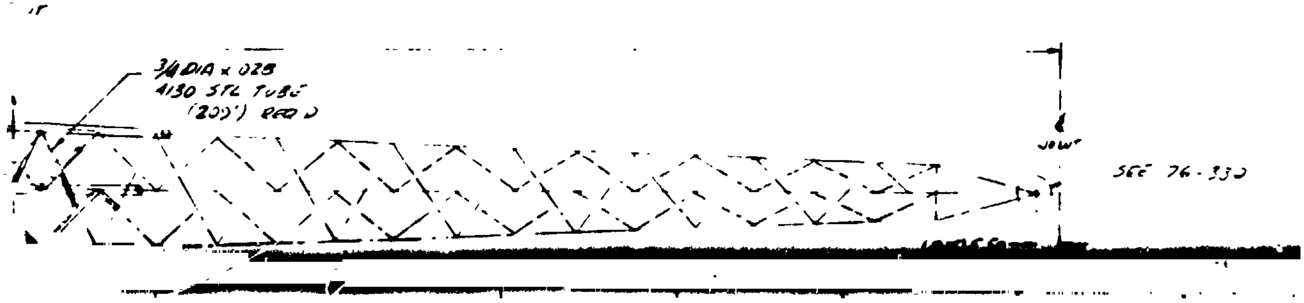


2.00
(REF)
3/4 DIA x .028
4130 STL TUBE (128) REQ'D
370.94
REQ'D
WT. 205#



3/4 DIA x .028
4130 STL TUBE
(200') REQ'D
SEE 76-330

OLDOUT FRAME 3

3	2	1

REVISIONS

ZONE	LTR	DESCRIPTION	DATE	APPROVED

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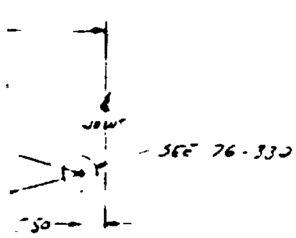

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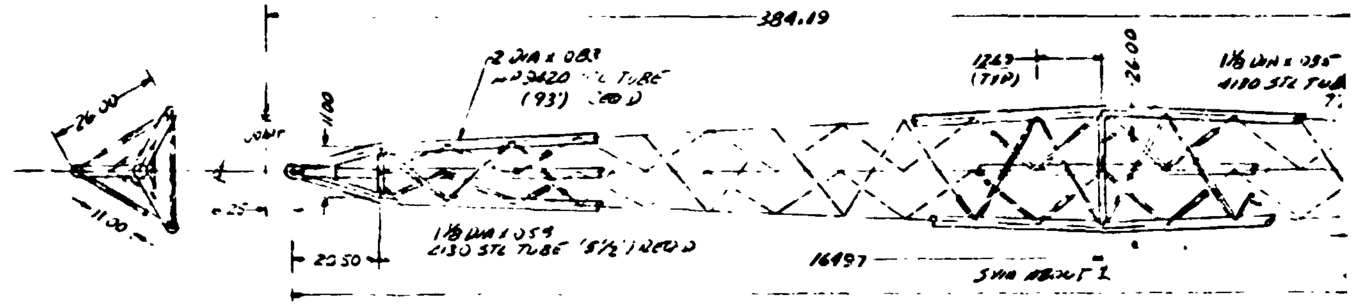
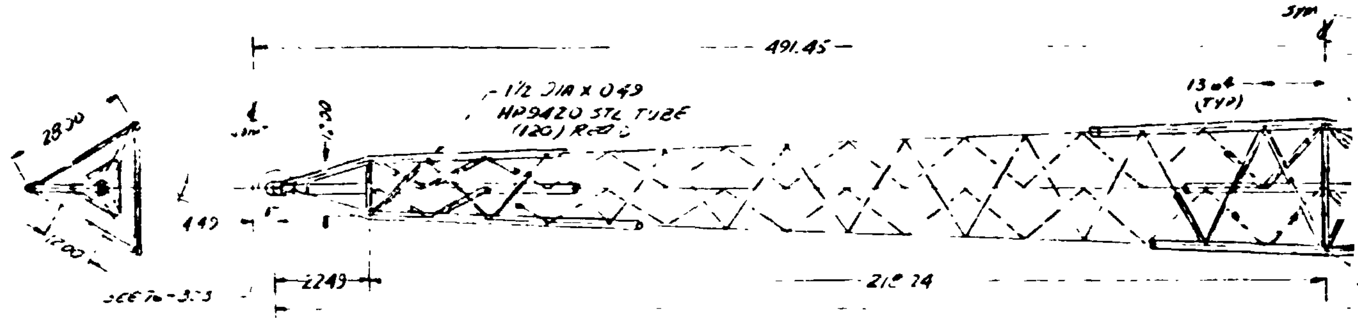
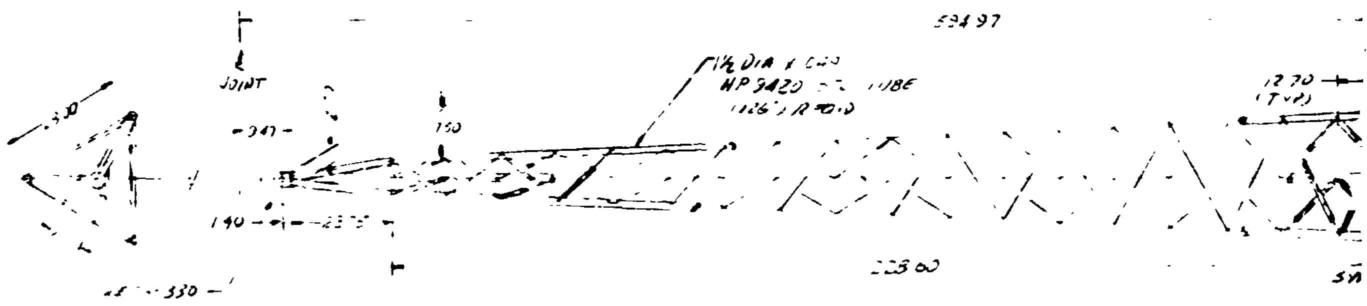


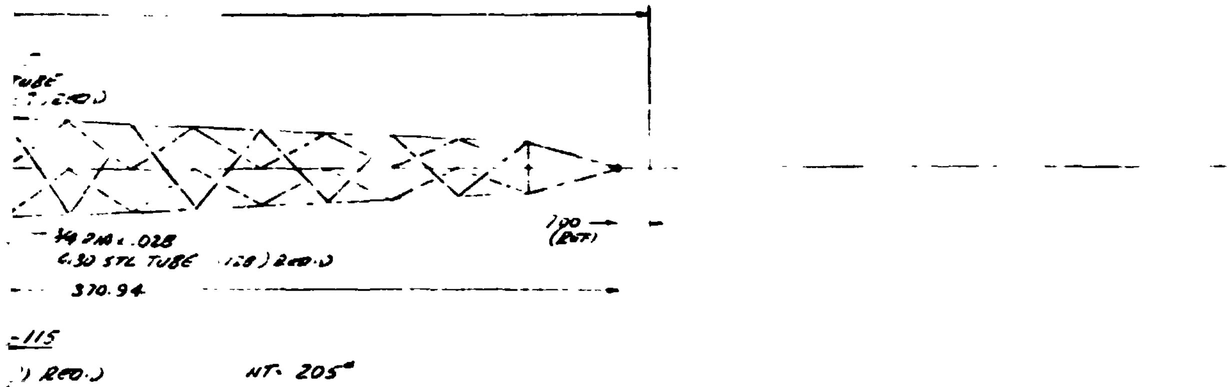
BI-POD BEAM -11
(2)

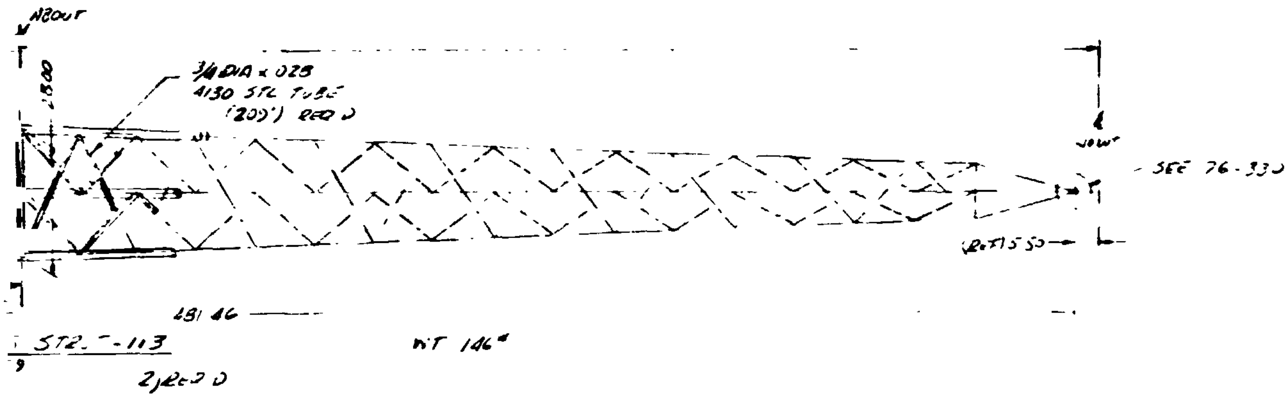


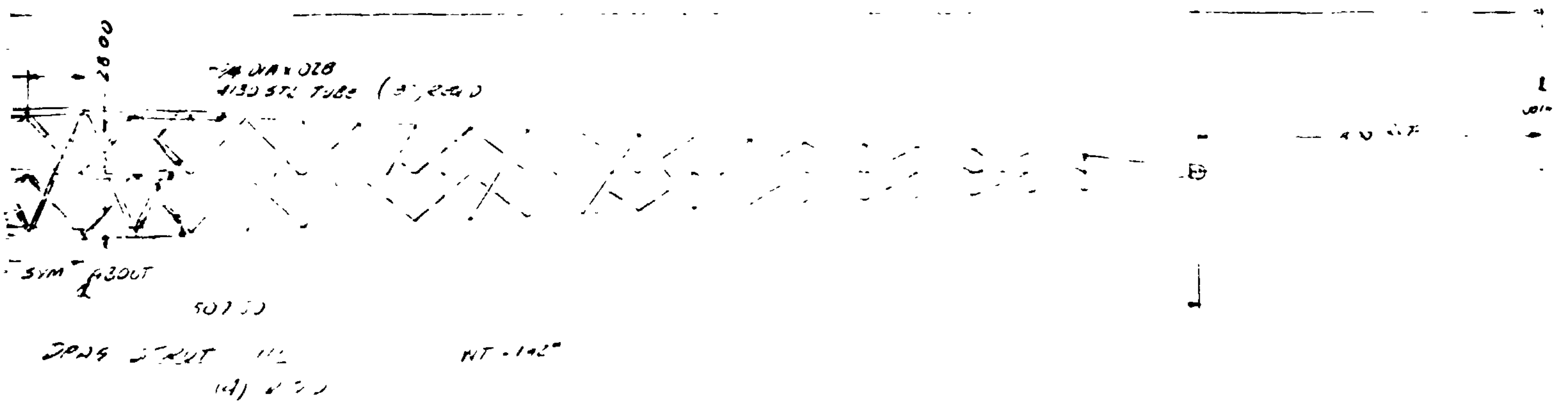
CROSS

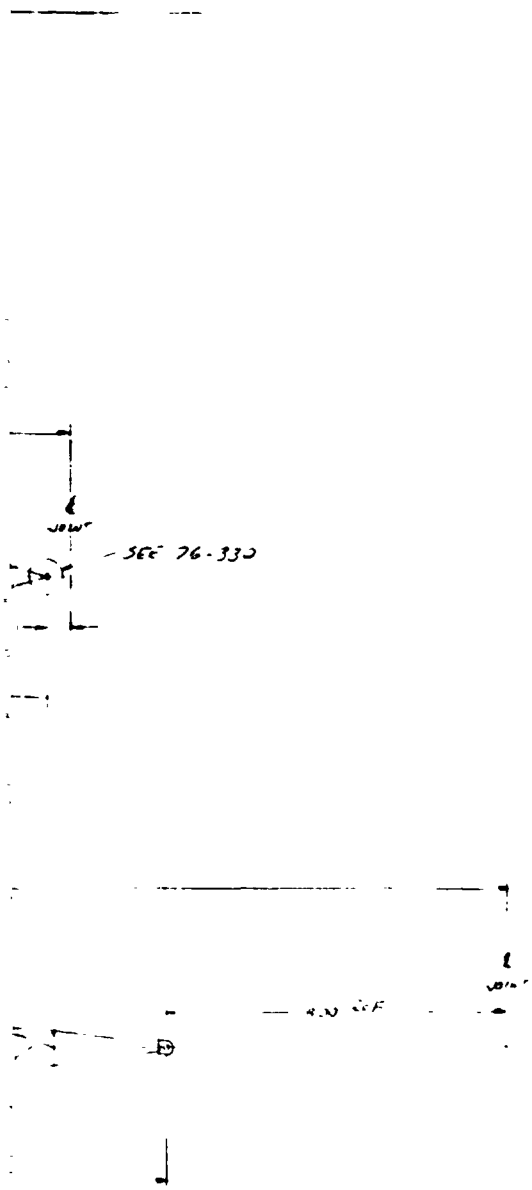


4


3/4 DIA x .028
4130 STL TUBE
370.94
-115
WT. 205#


ABOUT
3/4 DIA x .028
4130 STL TUBE
(209') REQ'D
SEE 76-330
481.46
STRUT -113
WT 146#
2 REQ'D


3/4 DIA x .028
4130 STL TUBE
SYM
WT - 142#



NOTE: [illegible] SIMILAR TO THOSE SHOWN ON SHT 1.

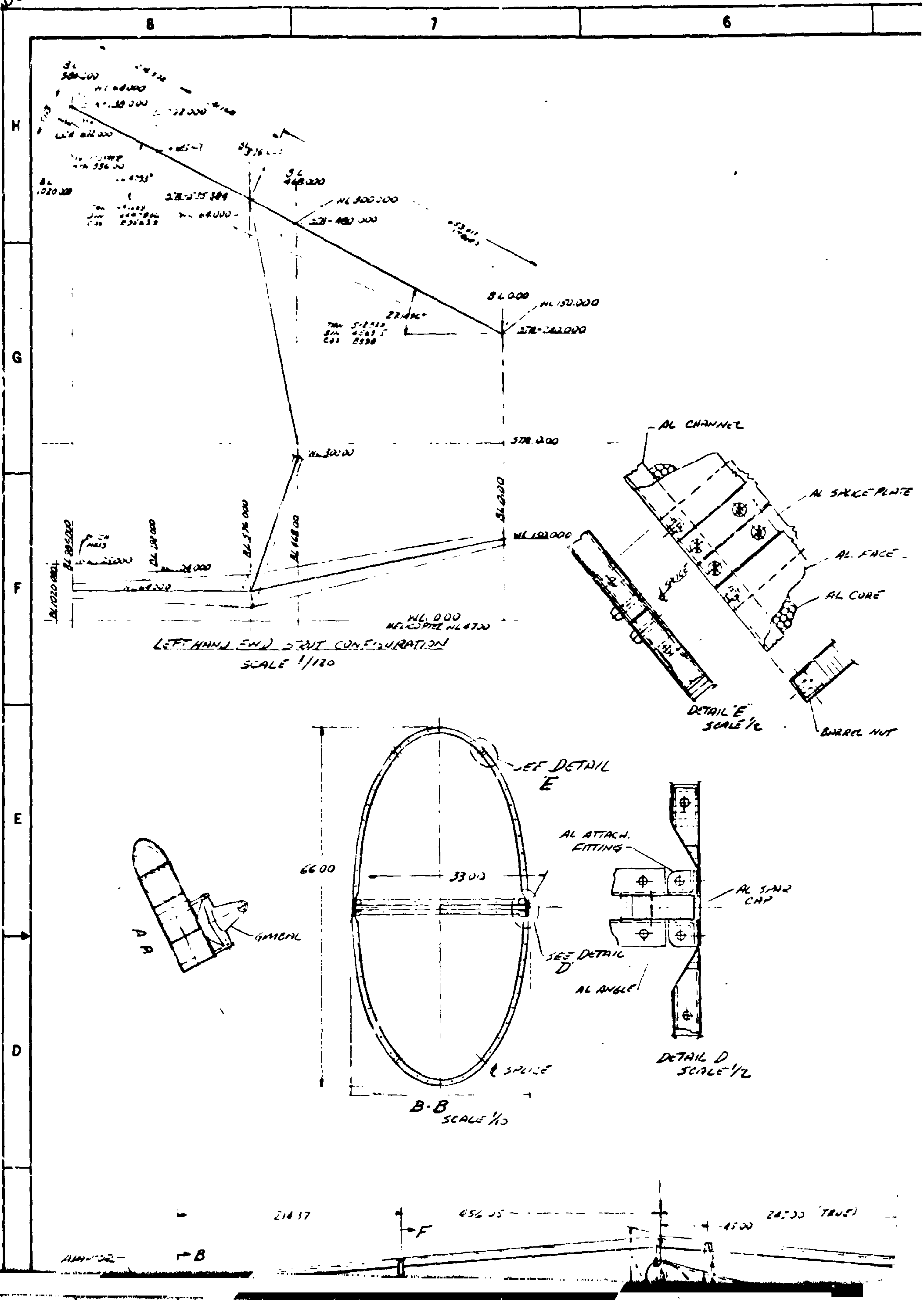
SIZE	CODE IDENT NO	DWG NO
E		

SCALE SHEET

FIGURE 5.8 STRUT ASSEMBLIES
(GOODYEAR DRAWING NO. 76-082)

NOTE: 1.0 in = 2.54×10^{-2} m

FOLDOUT FRAME

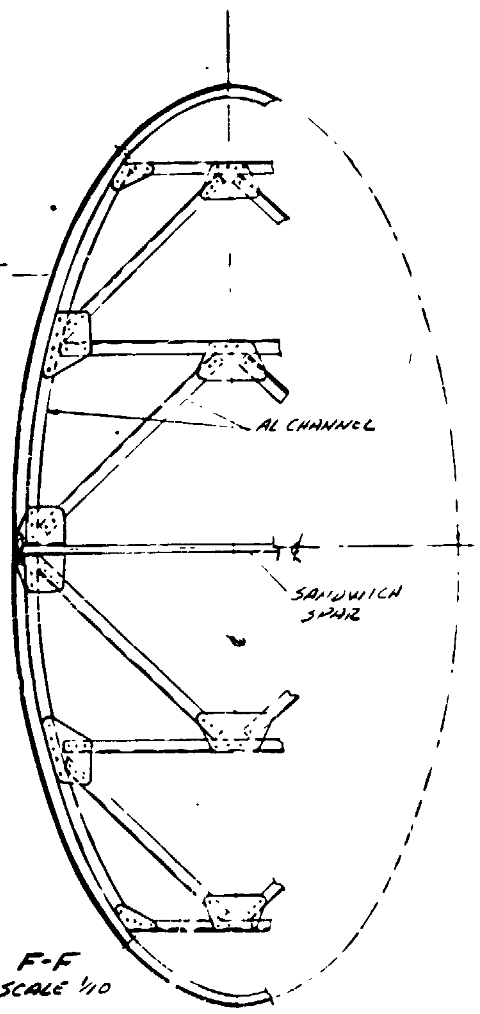


FOLDOUT 2

5	4	3	2

PLATE

SE SHEET



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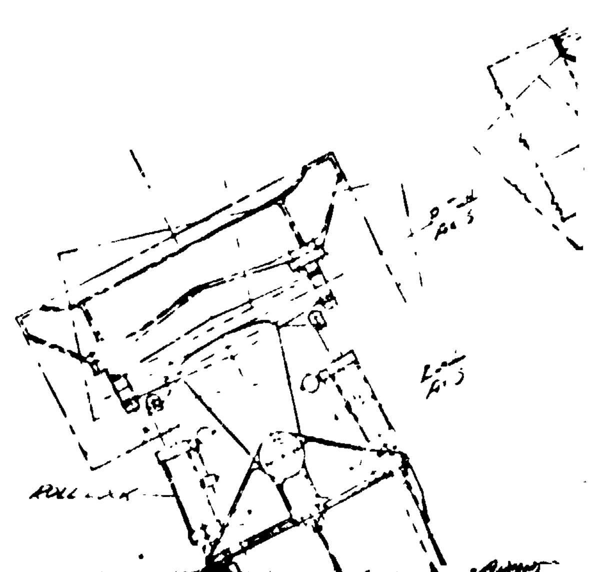

FOLDOUT FRAME 3

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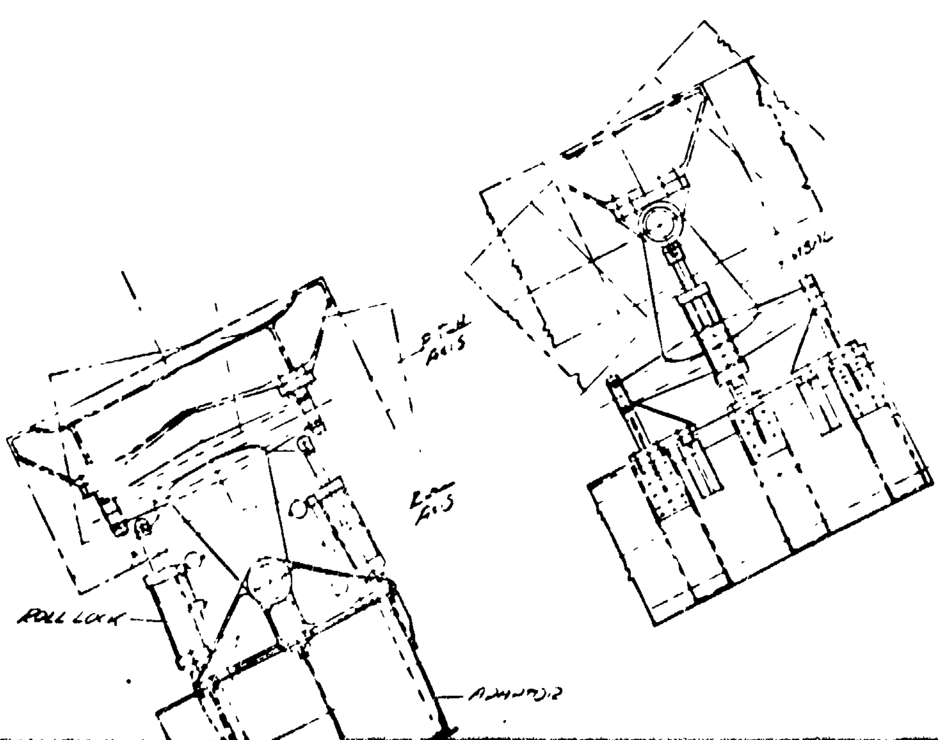

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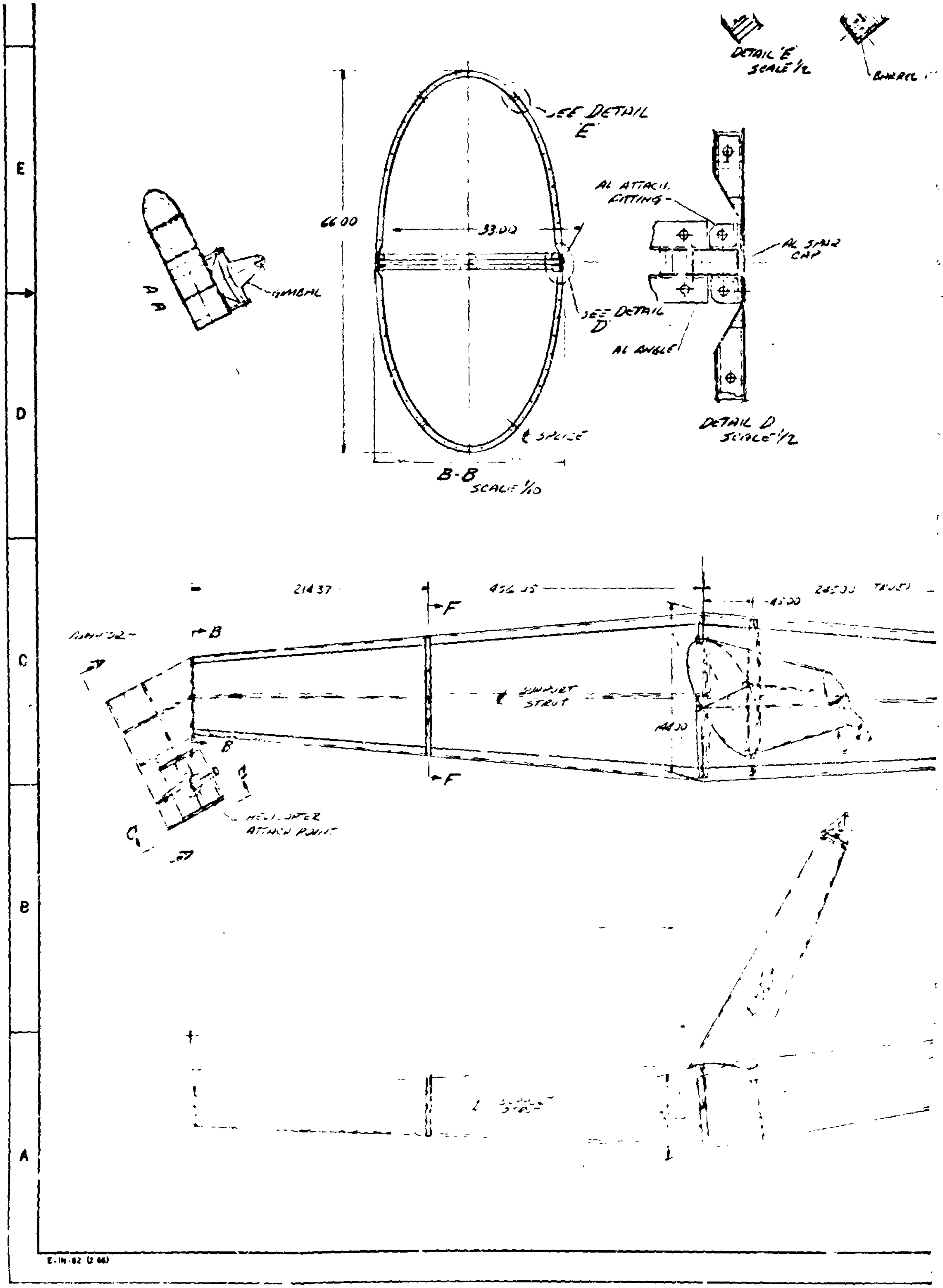

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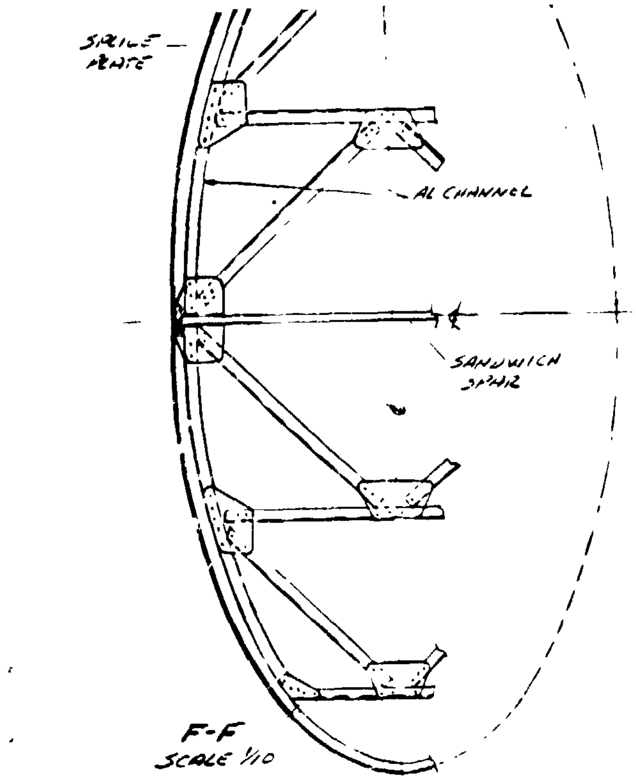

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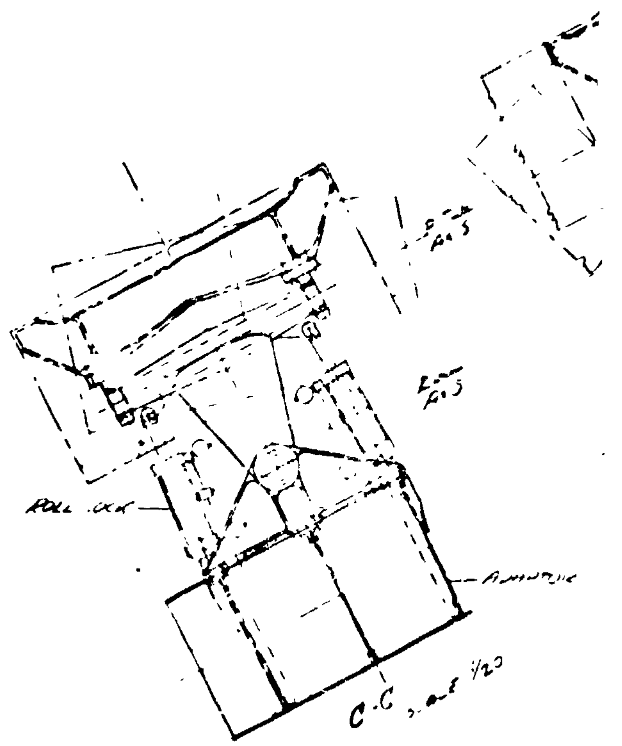

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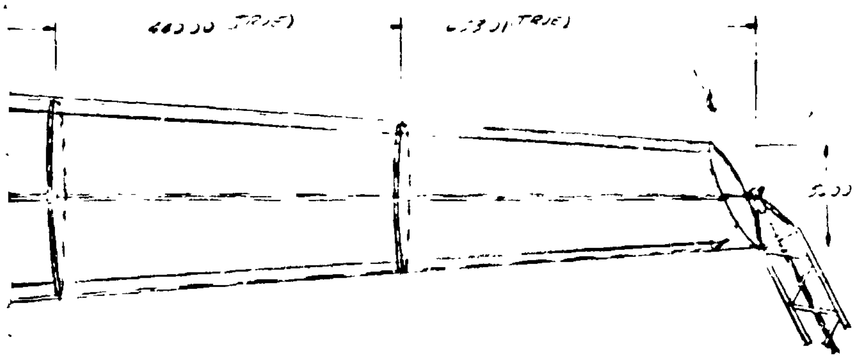

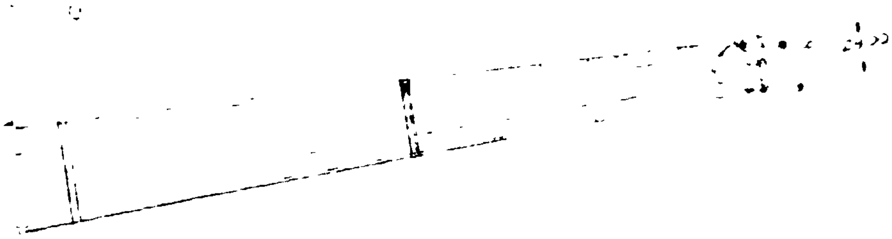

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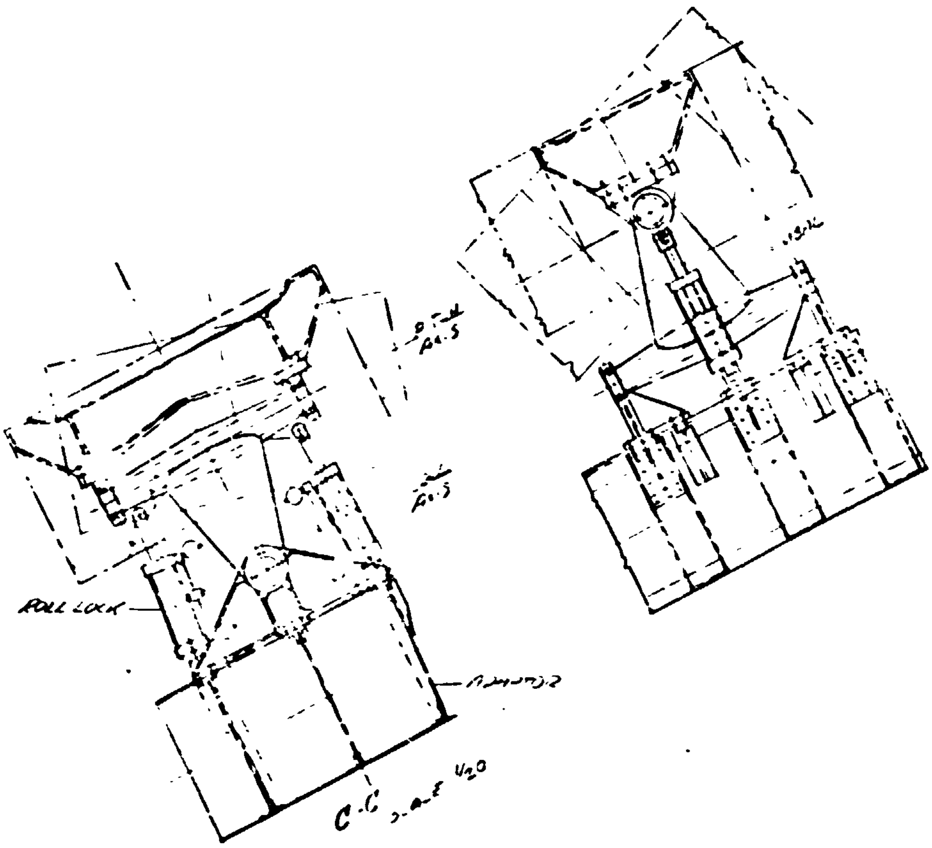

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FOLDOUT FRAME 6

The tail rotors of the aft helicopters are replaced with propellers and reoriented to provide sufficient propulsive force for forward flight and directional control at or near minimum gross weight. The tail rotors of the forward helicopters are used to provide side force for increasing the cross wind station keeping ability.

The vehicle is controlled through a FBW flight control system with the aft left helicopter serving as the command station. The FBW control ystem is similar to that developed during the Heavy Lift Helicopter (HLH) program which was successfully flown on a prototype basis in the tandem rotor CH-47 helicopter with over 300 hours of flight time accumulated. The HLH AFCS, PHS, and cargo-handling system have also been integrated into the Phase II HLA configuration.

The Phase II configuration permits a new center point mooring concept to be considered that minimizes mooring area and mooring mast requirements. Additional wind tunnel data (see Section 10) are required to permit a final assessment as to whether the concept can accommodate all mooring requirements of operational interest.

5.3 Fundamental Characteristics Requiring Special Consideration

5.3.1 Broad Based Suspension

During the preliminary studies of Phase I it became obvious that the heavy lifter concept introduces structural design conditions never before encountered in airship design. The basic reason for this is the fact that the maximum rotor forces available are in excess of the empty weight of the vehicle and are therefore capable of creating accelerations far in excess of previous experience. Furthermore, the very nature of the vehicle results in rotor locations which provide large moment arms and create the possibility of very large moments about all three axes being transmitted to the envelope. These considerations indicate a requirement for a broad based suspension system with an arrangement facilitating large rigging tensions in the cables so that no cables go slack in the most severe loadings.

5.3.2 Tail or No Tail

It has been obvious from the beginning that a dilemma exists with respect to the requirements for an empennage (tail). Provisions of a tail proportioned according to past design practice would greatly facilitate controllability of the airship in forward flight by providing a measure of aerodynamic stability. Such a tail would also make it possible to moor the airship on a mast (through a bow stiffening structure) as in past airship practice.

The disadvantage of the tail is that very high lateral forces and yawing moments are developed in the crosswind hovering condition creating very demanding requirements for control forces.

5.3.3 Interconnecting Structure

The large structure required to mount the helicopter rotors sufficient to preclude physical and aerodynamic interference between the rotors and the envelope and between adjacent rotors becomes an important consideration from the standpoint of structural integrity and inert weight. It was clear that considerable effort was justified toward defining a structurally sound, lightweight, interconnecting structure. Toward this end numerous design approaches were investigated.

5.4 Parametric Studies of the Interconnecting Structure

The initial parametric studies were directed toward achieving minimum weight in the interconnecting structure using a design approach which provided expectation of reasonable cost.

An H frame was envisioned consisting of a keel member and four outriggers. Design studies were conducted using conventional airplane fuselage (sheet, stringer, ring) approaches, rectangular and circular sandwich constructions and box truss arrangements. Of the approaches, the box truss arrangements appeared most promising from the standpoints of minimum weight, versatility, adaptability to providing the numerous "strong points" required and ease of construction with minimum tooling.

It was apparent that selecting the right size for the structural cross sections was an important consideration in achieving minimum weight and that optimum design of compression struts would be the prime factor in establishing the optimum size.

Details of the initial parametric studies of the interconnecting structure are provided in Appendix A of Book II of this volume of the report.

5.5 The Star Frame Interconnecting Structure

The parametric studies (see Appendix A of Book II of this volume of the report) of the interconnecting structure resulted in the realization that the optimum (minimum weight) design must have several features:

1) The very large planform dimensions require a large framework and leads to a structure with considerable depth for minimum weight.

2) In order to produce a relative simple structure strut members must be of a type that allows efficient design for low values of the structural index (P/L^2).

3) The importance of the "one engine out loading" requires a structure which is insensitive to keel torsion loadings. From this standpoint an "X" frame composed of beams running across from diagonally opposite helicopters would be ideal.

4) The final design must be capable of handling sizeable loads with six components at the helicopter mounts, provide convenient attachment points for a broad based suspension system, the payload sling, and tie down cables.

After several iterations, the "starframe" of Figure 5.9 evolved as a good solution for the interconnection structure. It appeared that this design constructed primarily from HP9420 steel girders would be able to carry the design loads with a total frame weight of less than 9072 kg (20,000 lbs).

This starframe arrangement did present some problems. The helicopters are mounted high off the ground representing a problem for access, servicing, and maintenance. A more serious problem was that the overall height of the airship precluded hangaring in all except the very largest hangars in the U.S.A. Another problem was the fact that the helicopters mounted on gimbals would dangle against the gimbal stops (at odd angles) when the vehicle was on the ground. Further, the large exposed structure created a severe aerodynamic drag penalty.

These considerations led to the "submerged starframe" design of Figure 5.10. Overall height is reduced to less than 36.58 m (120 ft) so that many existing hangars become potential refuges from severe weather. Helicopters are located on the ground when the vehicle is on the ground and sit on their own landing gears which are also capable of absorbing the overall vehicle landing shock. Only the outrigger support struts are exposed to the airstream and these are streamlined to reduce drag in forward flight.

The limited clearance between the helicopter rotor tips and the ground forces the helicopter support structure into a braced cantilever beam configuration which does produce a weight penalty. As will be shown, the weight penalty is on the order of 2268 kg (5000 lbs). It is judged that the advantages of this configuration outweigh the disadvantage of the weight penalty, and the "submerged starframe" is the configuration of choice for further evaluations.

5.6 Structural Design Conditions

Seven loading conditions (Table 5.1) have been selected for design of the interconnecting structure. The conditions were selected with the intent of providing an envelope of strength

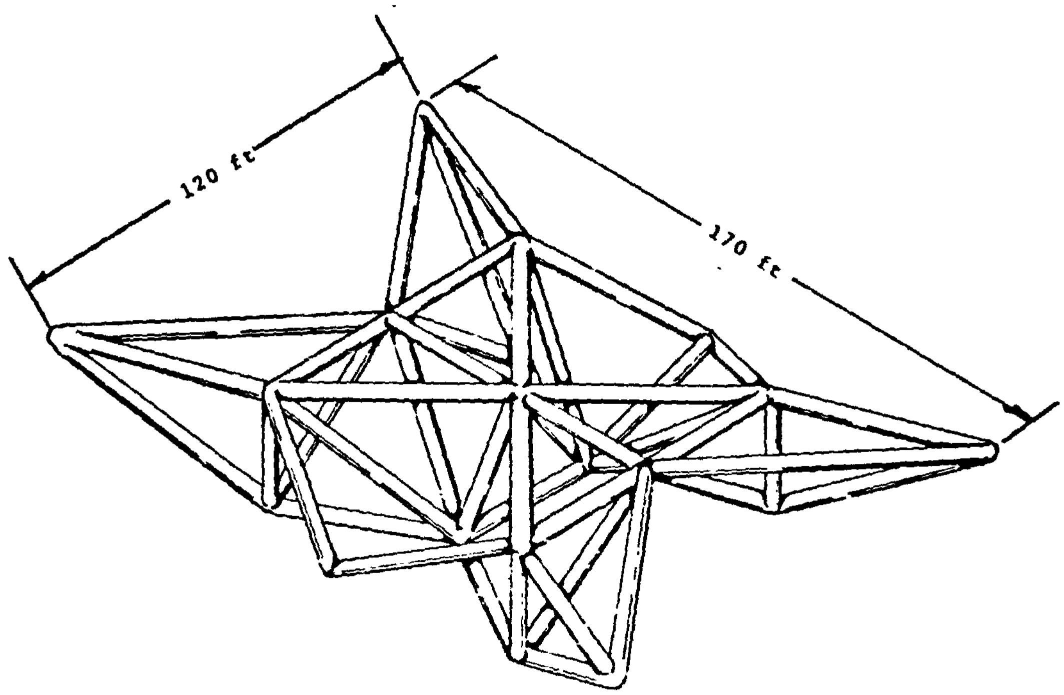

Figure 5.9 Star Frame Interconnecting Structure

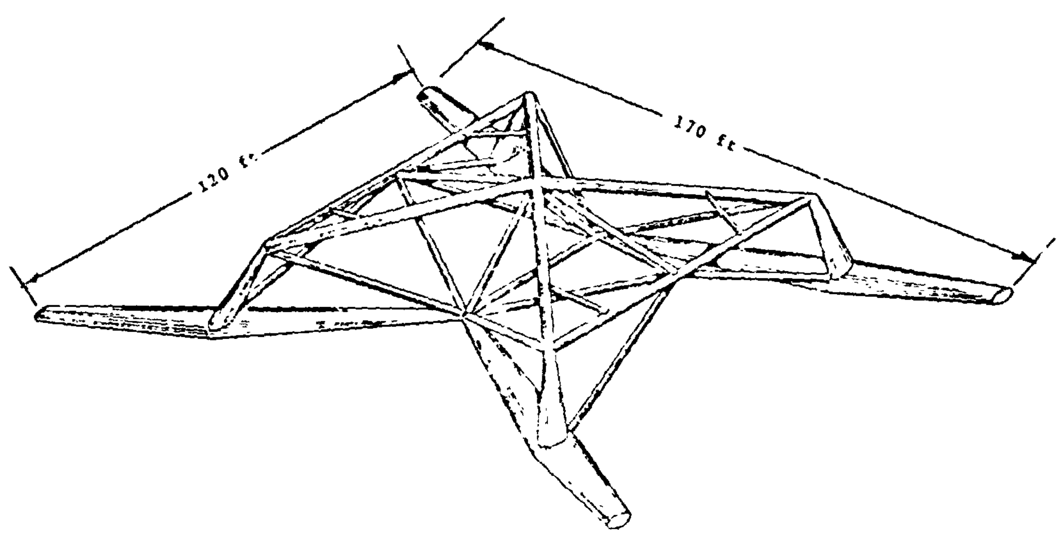

Figure 5.10 Submerged Star Frame Interconnecing Structure

TABLE 5.1. STRUCTURAL DESIGN CONDITIONS

1.1	HEAVY HOVER PLUS DYNAMIC COLLECTIVE PITCH
2.1	SAME AS 1.1 WITH ONE ENGINE OUT
3.1	HEAVY HOVER IN 30 KNOT CROSS WIND
4.1	MAXIMUM YAWING EFFORT
5.1	FOUR POINT LANDING 5 FT/SEC
6.2	TWO POINT LANDING 4 FT/SEC
7.1	CENTER POINT MOORING 65 MPH WIND BROADSIDE

NOTE: 1.0 knot = 5.144×10^{-1} m/s

1.0 ft/s = 3.048×10^{-1} m/s

1.0 mph = 4.47×10^{-1} m/s

sufficient for all normal flight and ground loads. The first four conditions are flight conditions and were selected on the basis of maximum vertical acceleration, maximum twist, maximum side load and maximum yawing effort. Two landing conditions are evaluated. The 4 point condition was selected as a possibly critical condition for outrigger and frame bending.

The two wheel landing condition was selected because it produces large negative bending loads on the outriggers and supporting structure as well as large frame twisting moments. The center point mooring condition is based on the mooring concept which supports the airship at the center of the planform at or near the ground line on a swivel fitting which allows the airship to orient itself broadside to the wind.

The loading conditions have been designated "1" through "7" with a digit after the decimal point used to indicate whether the helicopters are at minimum fuel (.1) or maximum fuel (.2). In all cases except one wheel landing the minimum fuel condition was found to be critical.

The response of the structural system to the dynamic loading conditions 1.1, 2.1, 5.1, and 6.2 have been analyzed on a simplified basis. This analysis is reported in Appendix B of Book II of this volume of the report along with the design loads resulting from the design conditions of Table 5.1.

5.7 Starframe Analysis

The analysis of the starframe is carried out by an integrated computer program (see Figure 5.11) which accepts the detail loads from Appendix B of Book II of this volume of the report, computes suspension cable loads, outrigger loads on frame, and loads in each member of the frame work. Some of the details of this procedure are provided in Appendix C of Book II of this volume of the report.

5.8 Outrigger Analysis

In this section is presented a preliminary analysis of the outrigger structure as represented in the design drawings to serve as a basis for the choice of dimensions and material thickness shown in the drawings. The loads applied at the gimbal

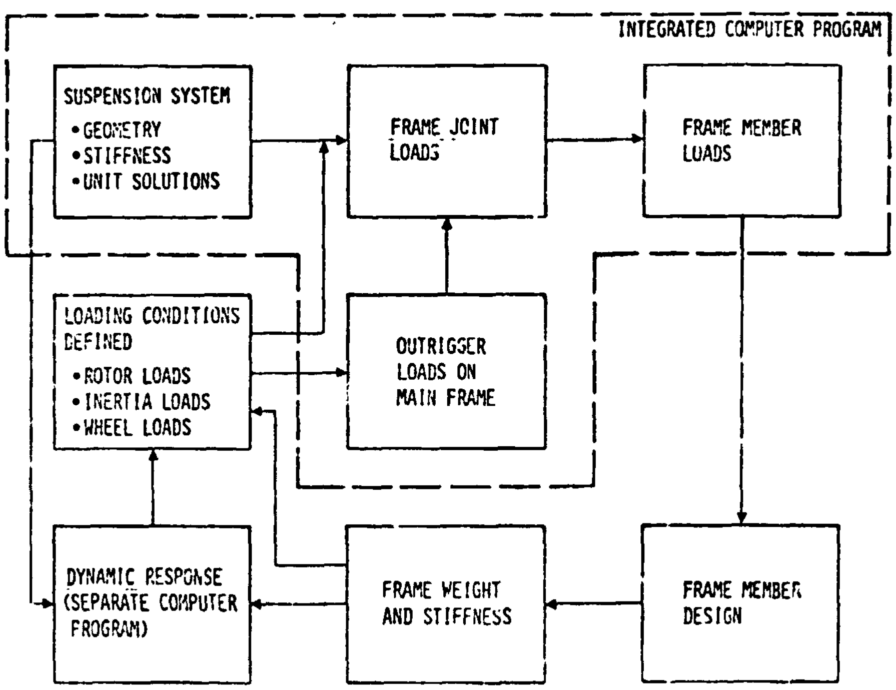

Figure 5.11 Frame Analysis Procedure

points in the seven design loading conditions selected for the interconnecting structure (see Appendix B of Book II of this volume of the report) are converted into shears and moments on three sections of the main struts. Section properties are developed. Bending and shear stresses are computed concurrently with determination of face sheet gages and spar cap areas. The detail analysis is confined to the main shell structure of the cantilever portion of the main strut. These results serve as a basis for estimates of the requirements and weights of the other components of the outrigger structure.

The four outriggers differ in geometry or design loads or both. The geometry differences between the forward and aft outriggers are necessary due to requirement to avoid physical interference with the main landing gear structures of the helicopters. Detail loads will be different for two reasons: (1) Main rotor torque is fundamentally antisymmetrical in nature on a basically symmetrical structure, (2) Tail rotor orientation is different between the fore and aft helicopters.

The analysis presented herein ignores these differences and treats the forward left outrigger structure as representative and the basis for strength requirements and weight estimates. A more thorough analysis in the future may show that the other three outriggers need to be different in specified details.

5.8.1 Structural Description

The general arrangement of the outriggers and their relationship to the starframe and helicopters is shown in Figure 5.1 (Drawing 76-069). Detail arrangement of the outrigger is shown in Figure 5.8 (Drawing 76-082). The outrigger consists of a main strut, lift strut, drag strut, and an adapter to support the gimbal from the main strut.

The main strut attaches to the end of the keel member of the starframe through a connection which permits 3 components of force but no moments. From this point it extends outward to the helicopter passing through the "elbow" joint where it is braced by the lift strut and the d g strut. The drag strut is a pin ended member and resists axial load only. The lift strut is connected to the starframe by a pin joint to provide 3 components of force but no moments. The lift strut is attached to the main outrigger strut through 2 fittings forming a hinge with its axis parallel to the fore and aft axis of the HLA.

The main strut is constructed as an aluminum alloy sandwich two-cell beam with an elliptical cross section and a center spar. Ribs are provided at the outboard end, elbow joint and inboard end to distribute concentrated loads. Additional ribs are provided at intermediate points to provide support to the shell and prevent general instability failures.

The lift strut is constructed similar to the main strut and experiences primarily an axial load and a chordwise shear/bending load.

The drag strut is a three-boom steel girder and is tabulated as a part of the starframe (see Table C-6 of Appendix C of Book II of this volume of the report). The adapter and gimbal on the end of the outrigger are not analyzed herein.

The details of the outrigger analysis are provided in Appendix D of Book II of this volume of the report.

5.9 Envelope Analysis

5.9.1 Design Criteria

The envelope pressure is selected so that (1) wrinkling will not occur under limit loads, and (2) excessive deformation will not take place under limit loads.

The fabric strength required for the various design conditions is dependent upon the frequency of occurrence of these conditions and the length of time the fabric is under stress in these conditions. Since in the design of an envelope the creep rupture strength is usually critical rather than the quick break strength of the fabric, the quick break strength is reduced by a factor which will guarantee adequate life of the structure.

This factor provides not only for creep rupture effects, but also nominal stress concentrations, wear and a scatter factor. The factors employed in this design are listed below.

1) A factor of 5.0 is used for conditions where the airship is under stress for long periods of time such as when masted out or in the hangar.

2) A factor of 4.5 is used for the limit design conditions which occur infrequently such as when masted out at 29.06 m/sec (65 MPH) or when in flight at 33.44 m/sec (65 Kts) and subject to a 15.24 m/sec (50 ft/sec) gust.

3) A factor of 4.0 is used for emergency conditions. This type of condition should not occur and if it does occur lasts only a brief period of time. An example would be if the airship is forced upward by gusts through its pressure height thereby opening the helium valves.

5.9.2 Determination of Required Envelope Pressure

In determining the required envelope pressure, the following critical design conditions were investigated:

1) Airship masted out

2) Landing

3) Flight - maneuver

4) Flight - gust

5) Flight - maximum yaw

Details of the Envelope Analysis are provided in Appendix E of Book II of this volume of the report.

5.9.3 Summary of Envelope Bending Moments and Pressure Requirements

The static, dynamic and aerodynamic moments determined for the above flight and landing design conditions (see Appendix E of Book II of this volume of the report) are summarized in Table 5.2. It is seen that the 7.87×10^5 m kg (5,697,285 ft lb) gust condition is the most critical.

TABLE 5.2 ENVELOPE BENDING MOMENTS

	Gust 50 FPS @ 65 Kt	Dynamic Collective	Four Point Landing	Max. Yaw
Static Moment (ft lb)	1,865,459	1,865,459	1,865,459	1,470,000
Aerodynamic Moment (ft lb)	3,831,827			
Dynamic Moment (ft lb)		1,028,481	1,028,481	1,410,000
Total Resultant Moment (ft lb)	5,697,285	2,893,940	2,893,940	2,036,900

NOTE: 1.0 fps = 3.048×10^{-1} m/s, 1.0 knot = 5.144×10^{-1} m/s, 1.0 ft lb = 1.382×10^{-1} m kg

The required pressure at the equator is

$$P_e = \frac{2M}{\pi R^3} = 1{,}132 \text{ N/m}^2 \text{ (4.55 in of } H_2O)$$

$$P_B^* = 973 \text{ N/m}^2 \text{ (3.91 in. of } H_2O) \text{ which was taken as } 995.4 \text{ N/m}^2 \text{ (4.0 in. of } H_2O) \text{ for convenience}$$

5.9.4 Fabric Stresses

From the calculations of the previous section it has been determined that the maximum operating pressure at the manometer at the bottom of the envelope will be 995.4 N/m^2 (4 in. of H_2O) in flight and 1244.2 N/m^2 (5 in. of H_2O) wh[illegible]ed.

When ascending through pressure height an additional increment of pressure of 373.3 N/m^2 (1.5 in. of H_2O) will be added to account for the pressure required to open the helium valves sufficient to permit the necessary rate of flow of helium to take place.

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*P_B is the bottom of the envelope which is typically the location at which envelope pressure is measured inflight.

All conditions involving maximum speed in flight will have an additional $\Delta q = 0.16q$ negative external pressure added to the internal pressure to account for the aerodynamic pressures surrounding the envelope.

Fabric stresses for the following conditions were investigated:

1) Masted at 29.06 m/s (65 MPH)
2) Masted - calm weather
3) Gust in 30° pitched flight
4) Ascent through pressure height

The creep rupture factors employed for the above conditions were 4.5, 5.0, 4.5 and 4.0 respectively. The maximum stresses for these critical conditions are summarized in Table 5.3.

From Table 5.3 it is seen that the masted condition at 29.06 ms/ (65 MPH) condition is critical and will require a fabric having a strength of 11,478 kg/m (643 lbs/in).

TABLE 5.3 REQUIRED FABRIC STRENGTHS

	Pressure at the Bottom of the Envelope (in. of H_2O)	Δq (in. of H_2O)	Total Pressure (in. of H_2O)	Attitude	Max. Stress (lb/in)	Factor	Required Strength (lb/in)
Masted at 65 K	5.0	0	5.0	Horizontal	143	4.5	643
Masted - Calm Weather	5.0	0	5.0	Horizontal	115.9	5.0	580
Gust	4.0	0.44	4.44	30° Pitch	112.32	4.5	501
Ascent through Pressure Ht.	5.5	0.44	5.94	30° Pitch	146.12	4.0	580

NOTE: 1.0 kt = 5.144×10^{-1} m/s, 1.0 in. of H_2O = 2.49×10^{2} N/m^2, 1.0 lb/in = 17.85 kg/m

5.9.5 Fabric Weight, Envelope

The fabric will consist of a neoprene coated Type 68 Dacron cloth having the construction given below.

Alum. Coat	0.041 kg/m^2	(1.20 oz/yd^2)
Neoprene	0.034 kg/m^2	(1.0 oz/yd^2)
Bias Ply (Dacron)	0.178 kg/m^2	(5.25 oz/yd^2)
Neoprene	0.271 kg/m^2	(8.00 oz/yd^2)
Straight Ply (Dacron)	0.297 kg/m^2	(8.75 oz/yd^2)
Neoprene	0.051 kg/m^2	(1.5 oz/yd^2)
	0.87 kg/m^2	(25.7 oz/yd^2)

5.9.6 Envelope Weight

Employing a 0.87 kg/m^2 (25.7 oz/yd^2) fabric the weight of the envelope will be 16,489 kg (36,360 lbs), the breakdown of this weight is given in Table 5.4.

TABLE 5.4 ENVELOPE WEIGHT

[illegible]lope	[illegible]1,160
Ballonets	2,500
Pressure System	3,690
Miscellaneous Envelope & Car Fairing and Air Lines	2,476
Internal Suspension - Curtain (Dacron)	1,360
Internal Suspension - Cables (Steel)	2,[illegible]70
External Suspension	2,904
TOTAL	36,360 lbs

NOTE: 1.0 lb = 4.535×10^{-1} kg

5.9.7 Envelope Shape Analysis

5.9.7.1 Center Point Mooring Condition

The most severe distortion of the envelope cross section occurs in the center point mooring condition which is evaluated for a broadside wind of 29.06 m/s (65 MPH). The capability of being able to moor out in a 65 MPH condition will result in at most an infrequent necessity to select an alternate mooring site.

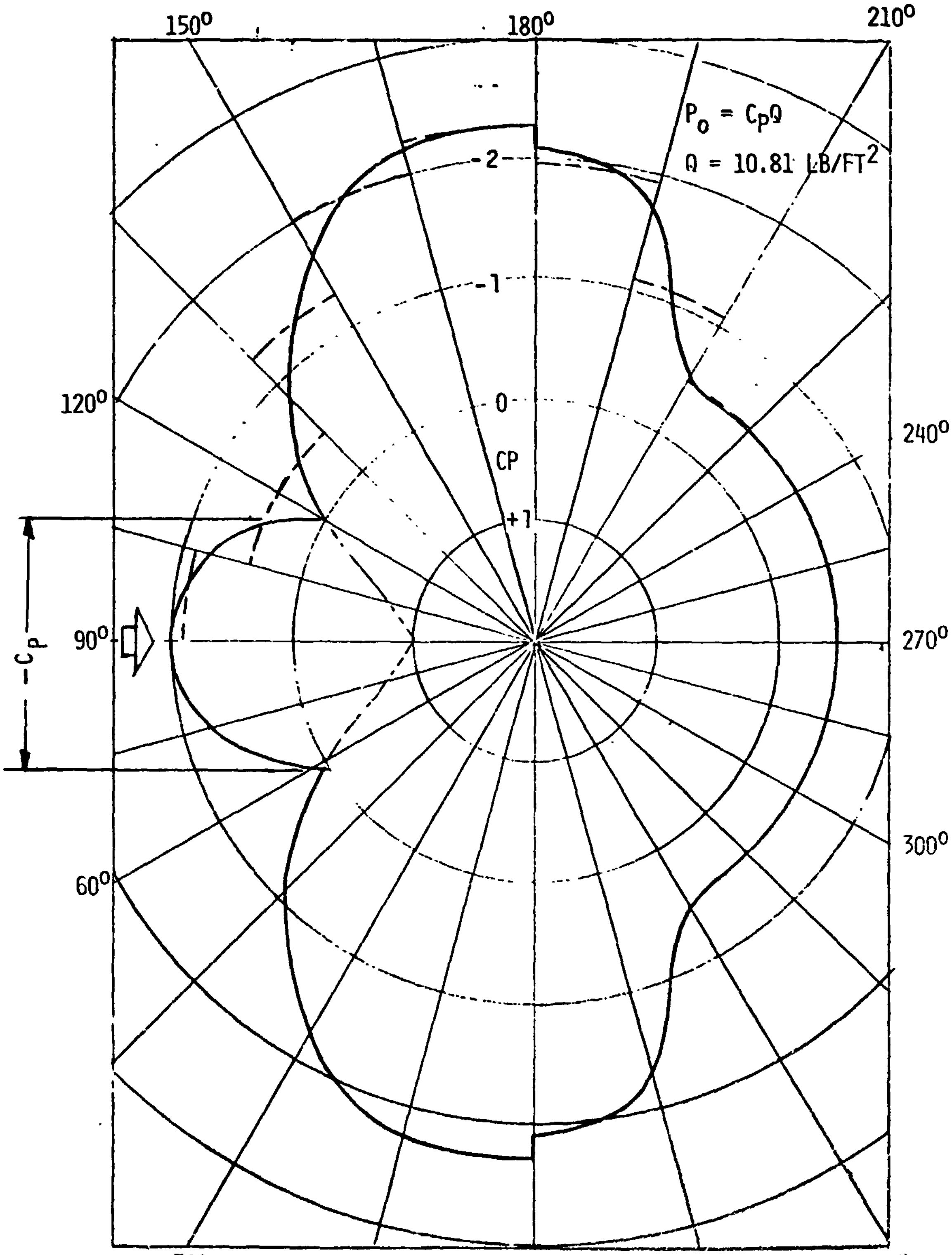
The aerodynamic pressure distribution used to calculate the inertial loads on the hull is based on data in Reference 6 and is shown in Figure 5.12 along with the simulated distribution used in the analysis. The distortion analysis considers the envelope fabric to attach directly (and continuously) to the shoulder beams of the starframe. The constraints of the internal catenary system are also a part of the analysis. At the start of the analysis, it was anticipated that the internal suspension system might go slack on the windward side which could only be prevented by rigging deep valleys into the envelope at the attachment line of the internal curtains. For this reason the calculations were made for several values of the internal system radius. It turns out that the cross wind pressure distribution produces large suction loads on the top of the envelope and there is no tendency for the internal system to go slack. The choice of radius "a" therefore was based more on the symmetrical buoyant lift condition, the envelope cross section shape and associated distribution of vertical loads in the rigging condition.

The details of the envelope distortion analysis are included in Appendix E of Book II of this volume of the report.

5.9.7.2 Symmetrical Loads

Several related studies were made to determine the appropriate length (cables plus curtains) of the internal suspension system, the distribution of vertical loads as a function of the suspension geometry and variation in the distribution resulting from changing super pressure. The change in the location of the center of gravity of the envelope cross section from the rigging condition

C.2



NOTE: 1.0 lb/ft^2 = 47.88 N/m^2

Figure 5.12 Pressure Distribution Center Point Mooring of 45,350 kg (100,000 lbs net lift) to a no-lift condition (air inflated shape) was claculated as a simple approximation of the stiffness of the suspension system for dynamic vertical loads.

For these calculations, the external catenary system was configured to attach to the envelope ±15° from the nominal 45° location of the shoulder beams of the interconnecting framework. The shoulder beam was assumed to provide a standoff between the suspension attachment point and the inner edge of 1.15 m (3.71 ft) which creates a geometry which does not distort the circular cross section of the envelope in the unloaded condition (see Figure 5.13).

The cross section shape calculations were carried out similar to the procedure described in Section 5.9.7.1 with the constraints adjusted to satisfy the requirements of symmetry at top center. Details of the envelope shape analysis for the symmetrical loads are presented in Appendix E of Book II of this volume of the report.

5.10 Phase II HLA Weight Summary

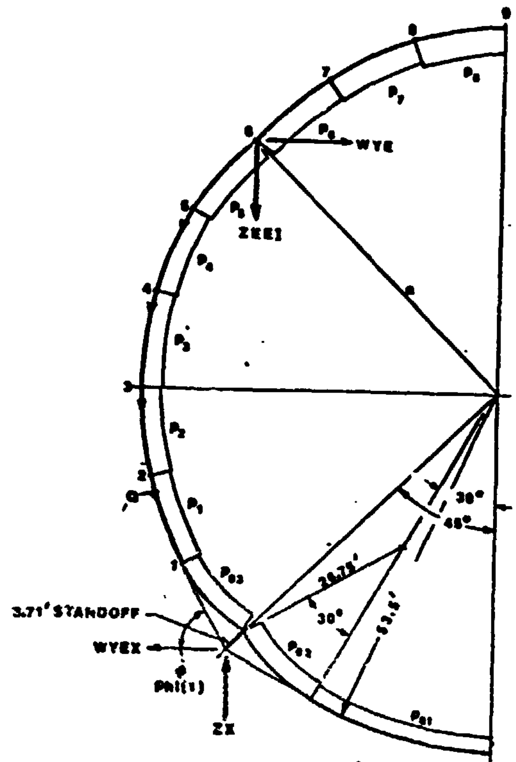
The estimated weight of the Phase II HLA is provided in Table 5.5. This weight summary is based on the component analysis and design drawings presented in Sections 5.2 through 5.8; actual weight statements for the CH-54B with appropriate modifications*; available weight data for the HLH components; estimated electrical cabling and cabling support requirements for connecting all helicopters to the command helicopter; and estimated vehicle sensor requirements.

5.11 Flight Dynamics Analysis

5.11.1 General

One factor leading to Goodyear's Phase I recommendation of the HLA configuration was the judgment that it possessed far fewer aerodynamic uncertainties than other heavy lift concepts

* Appendix F of Book II of this volume of the report provides a detailed weight statement for the helicopters.



NOTE: 1.0 ft = 3.048×10^{-1} m

Figure 5.13 Load Diagram - Envelope Symmetrical Loads and Rigging Load Shape

TABLE 5.5 ESTIMATED EMPTY WEIGHT OF PHASE II HLA

WEIGHT EMPTY (Pounds)			148,070
Helicopters with Modifications (4)*		83000	
Envelope Group		36360	
Envelope	21160		
Ballonets	2500		
Pressure System	3690		
Misc. Envelope and Fairings	2476		
Internal Suspension Curtains	1360		
Internal Suspension Cables	2270		
External Suspension	2904		
Interconnecting Structure		27500	
Internal Starframe (includes Drag Strut)	7500		
Support and Lift Struts	20000		
Precision Hover Sensor		540	
Automatic Flight Control System Electronics		20	
Fly-By-Wire Control System		600	
Electronics	100		
Interconnecting Cabling and Supports	500		
Vehicle Sensors and Cabling		50	

* Includes adaptor to support strut

NOTE: 1.0 lb = 4.54×10^{-1} kg

combining buoyant and rotor lift. The uncertainty with respect to the Phase II concept was one of whether any large interference problems would be experienced in the combination of large rotors in close proximity to a large hull. Tandem rotor helicopter experience indicates that rotor-rotor interference will not be a problem. The corporately sponsored wind tunnel and flight dynamics computer simulation development efforts have shown the HLA concept feasible and capable of hovering in a considerable crosswind. These efforts have indicated, however, that attainment of current crosswind hovering goals will require further wind tunnel efforts to refine the current Phase II configuration.

5.11.2 Exploratory Wind Tunnel Investigation

NASA Ames provided the use of the ARC 7 x 10-foot wind tunnel facility for this initial exploration effort which was directed principally at:

1) The effect of hull proximity on rotor performance for flight and hover conditions in ground and out of ground effect.

2) The effect of one rotor's operation on the performance of the remaining rotors.

3) The effect of the rotor's operation on the basic aerodynamic characteristics of the hull.

4) Definition, where Reynolds number and balance sensitivity permit, of the basic static aerodynamic characteristics of the vehicle.

The model (see Figure 5.14) was designed, fabricated, and tested for Goodyear by Nielsen Engineering and REsearch (NEAR), Inc. NEAR also analyzed the test data and assisted in the application of that data to the Goodyear 6 DOF flight dynamics computer simulation. Book III of this volume of the report delineates in detail pertinent aspects of the wind tunnel effort including

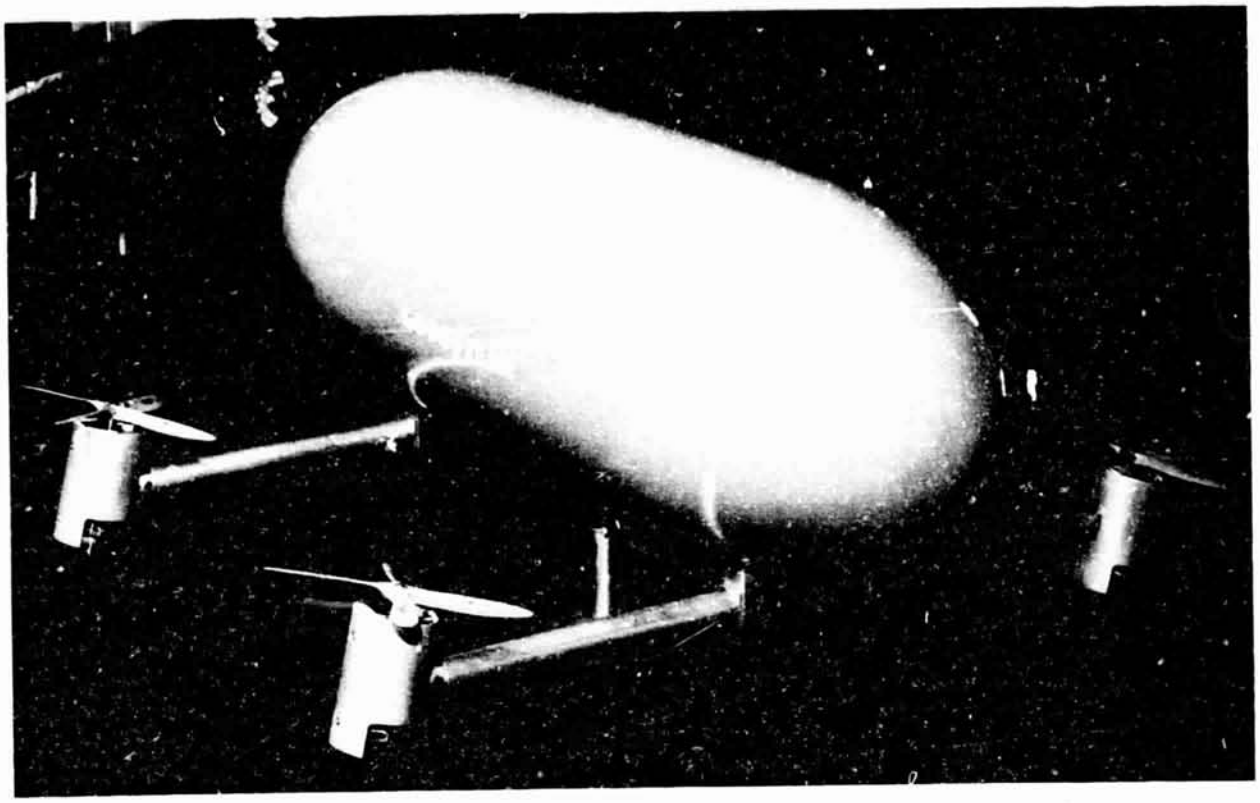


Figure 5.14 Powered Model in ARC 7 x 10-Foot Wind Tunnel Facility in Presence of Ground Plane

reduced data, analysis of the data, recommendations resulting from the testing, etc.

Figure 5.14 shows the model installed in the ARC 7 x 10-foot facility in the presence of the ground plane with the rotors tilted in roll as in the case of a crosswind hovering condition. The model provides the following general features:

1) Rotor location can be varied with two axial and two lateral positions possible.

2) Rotor plane can be rolled as would be necessary when hovering in a crosswind condition.

3) The model is capable of accepting a vertical tail surface.

4) The model mounting provisions were such that the height above the ground plane could be varied.

5) Model instrumentation included:

   a) A six component main balance

b) A six component balance in each of the four outriggers

c) A magnetic pickoff on each motor which through a frequency to voltage convertor was used to measure propeller rpm.

The following represent the most significant conclusions from the exploratory testing:

1) No appreciable interference effects in forward flight were observed.

2) No appreciable interference effects were observed for angles of side slip less than 60 degrees (at α = 0 degrees).

3) In general, the rotor-rotor interference was negligible and the effect of the hull on the rotors small.

4) The most significant adverse effect observed was the considerable increase in crossflow drag of the hull as a result of the operation of all four rotors. The interference effects that were observed are a strong function of rotor placement with the effect decreasing as the rotors are moved outboard. Fore and aft displacement of the rotors resutled in no appreciable change in the observed interference effects.

5) The modification of the flow field around the hull by the rotors may be a usable phenomenon in controlling the vehicle. More testing, however, is necessary to confirm this.

A hovering interference model was developed from the wind tunnel data and included in the 6 DOF flight dynamics simulation. The simulation results which are discussed in detail subsequently, indicate that the maximum crosswind hovering capability of the vehicle as presently configured at maximum gross weight is on the order of 10.29 m/sec (20 knots). As discussed below, several configurational changes are possible for increasing this crosswind hovering capability. It is also considered appropriate that the 15.43 m/sec (30 knot) crosswind requirement be reconsidered in view of alternative operational approaches which are reviewed in Section 8.0 of this report. With reasonably simple modification, the flight dynamics simulation can be used to evaluate the merits of various alternatives.

Configurational changes that are recommended in Section 10 of this document for reducing the interference effects observed in the testing of the HLA include:

1) Moving the rotors farther outboard
2) Increasing the envelope fineness ratio while retaining the rotors in the current position
3) Changing the cross section of the hull to an elliptical shape
4) Vertical displacement of the rotor plane

The above changes should be investigated in further exploratory wind tunnel testing as recommended in Section 10. In the final analysis, however, the recommended flight research vehicle will be required to establish the actual crosswind capability needed and to assess the viability of operational alternatives.

5.11.3 Six Degree of Freedom Flight Dynamics Hybrid Computer Simulation

5.11.3.1 General

A 6 DOF simultion has been developed using the Goodyear hybrid computer facility to assess the flight dynamic characteristics of the HLA. Other uses of the simulation included:

1) Synthesis of overall control system requirements
2) Synthesis of the fly-by-wire control laws and autopilot characteristics

3) Verification that the control laws developed for interface with the AFCS and PHS modes are compatible with manual modes.

4) Precision hover mode accuracy

5.11.3.2 Description of Simulation

5.11.3.2.1 General

Figure 5.15 indicates the elements involved in the simulation and their respective location in the hybrid setup. The HLA gust model, control laws and autopilots, which are described in Sections 5.11.3.2.2.3, 5.11.3.2.2.4, and 5.11.3.2.2.5 respectively, are programmed on the EAI 7800 analog computer. The analog is linked, through an EAI 8831 hybrid interface system to a Xerox Sigma 9 digital computer which contains the equations of motion, airship and helicopter non-linear aerodynamics, and the crosswind hover interference model. In addition, the coordinate transformation to resolve body velocities into inertial velocities are performed on the Sigma 9. The HLA was modeled in six degrees of

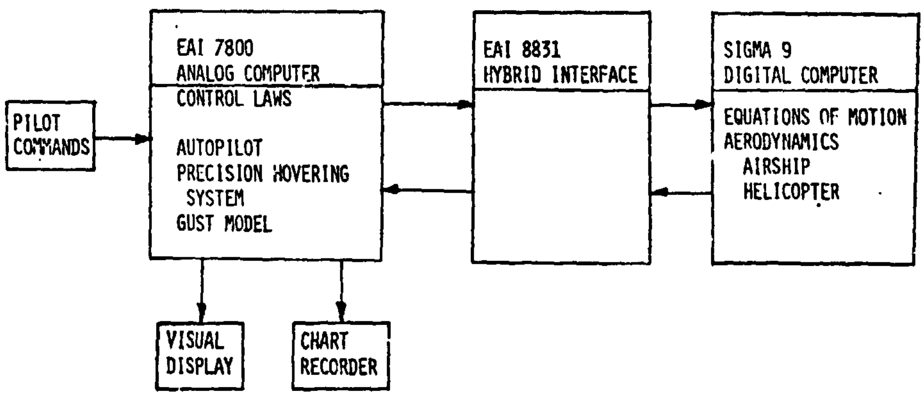


Figure 5.15 HLA Six DOF Hybrid Computer Flight Dynamics Simulation

freedom in the body axis for several conditions including: (1) maximum gross weight; (2) minimum gross weight; and Items (1) and (2) with and without tail and in and out of ground effect.

The HLA was flown using an AFCS for cruise and hover modes. In addition the simulated HLA can be flown manually by stick inputs to the analog computer. However, the simulation was generally not "flown" this way because of the inconsistency of results due to the pilot's lack of repeatability. The simulation operates in real time to provide realistic flight responses for flight inputs.

All the results are displayed as an output of the analog computer on an oscillograph or on a visual display which shows a side view or plan view (either can be selected) to aid interpretation of the results. Each view presents three degrees of freedom. The side view shows X, Z and θ and the plan view shows X, Y and ψ. The roll attitude ϕ is not shown but this is of minor importance since the metacentric moment is a stabilizing force in roll and roll does not directly effect the HLA flight path. In general, roll attitude can be left unattended by the HLA pilot except for the critical maneuvers of landing and precision hover very close to the ground, where roll attitude is important.

Currently the simulation considers the helicopters to be rigidly attached to the interconnecting structure with cyclic pitch control being the only mechanism for changing the direction of the main rotor thrust vector. The application of cyclic pitch is rate limited at 100% per second and a rotor time lage of 0.3 second is also included. Inclusion of the kinematics of the individual helicopters will not alter the rate at which the thrust vector can be redirected in pitch and thus the directional control response will not be affected. It is anticipated that any deterioration of the rate at which the thrust vector can be redirected in roll will be insignificant and as a result the precision hovering response currently being obtained will not be perceivably altered. It is anticipated that the simulation will be used to define the required roll response (roll servo specifications) to provide satisfactory hover response.

5.11.3.2.2 HLA Aerodynamic Characteristics

5.11.3.2.2.1 Static Characteristics

The aerodynamic characteristics used in the flight dynamics simulation were developed during a corporately sponsored activity associated with the overall development of the simulation capability. The static aerodynamic characteristics of the hull were developed from the HLA wind tunnel data (see Book III of this volume of the report) and the Goodyear LTA Aerodynamics Handbook. The axial drag characteristics of the helicopters were developed from References 1 and 7. The crossflow drag characteristics of the support, lift, and drag struts were developed from Reference 8 data.

The crosswind hovering requirement and the center point mooring concept (which results in the hull stabilizing broadside to the wind) represent unique attitude requirements for an airship hull. Historically, airships have flown at only small angles of α and β. Mooring has historically been about some forward point on the ship, generally the nose. Mooring analyses have historically considered approximately a 12° change in direction of the relative wind (with which the airship is originally aligned) in calculating forces, etc. As a result there has historically, in general been a lack of interest in the aerodynamics of airships at large angles of sideslip.

Of particular interest in the crosswind hovering and center point mooring concept is the side force coefficient as a function of angle of sideslip. With the current degree of rotor induced side force on the hull, the hull side force coefficient is not extremely significant in defining the crosswind hover capability. However, as the interference effects are reduced during future development efforts via the approaches discussed in Section 5.11.2 the hull side force coefficient will become a more significant factor in the crosswind hovering capability.

References 9 and 10 present wind tunnel data for a 1/40 scale model of the airship Akron at a Reynolds number of 2.9×10^6 based on the maximum diameter of the model. Side force data for the complete airship (hull plus tail) in the moored condition is presented as a function of angle of sideslip up to 90°. Reference 9 presents data for this same model at essentially the same Reynolds number both for the hull alone and the hull with tail without the influence of a ground plane. Analysis of this data would suggest that the HLA hull alone side force data (see Book III of this volume of the report) is high compared to values that will be obtained at Reynolds numbers closer to the full-scale value of 31.9×10^6 (which is based on a full-scale hull diameter of 107 feet and a crosswind velocity of of 30 knots). The HLA hull side force data were obtained at Reynolds numbers up to 0.56×10^6 based on hull model diameter. Testing of the hull in the recommended ARC 40 x 80-foot facility will permit Reynolds numbers up to approximately 14×10^6 based on maximum model diameter to be obtained. In addition, the recommended testing in the ARC 40 x 80-foot facility will permit pressure distributions to be obtained in the crosswind atti when moored. There is currently no directly applicable dat. this area. It is known that the pressure distribution data of Reference 6, which was used in the analysis of the envelope when moored, results in an optimistic picture of the fabric loads since the pressure distribution does not include the effects of a ground plane. In order to make a final evaluation relative to the center point mooring concept, the recommended testing in the ARC 40 x 80-foot facility is essential. As discussed in Section 8.0 there are several alternatives to the center point mooring concept which will be considered should the center point mooring concept ultimately be found to result in too severe a fabric weight penalty.

Conservatively, the crosswind hovering simulation studies, which are presented subsequently, use the large side-force coefficients obtained in the recent HLA wind tunnel investigation (see Book III of this volume of the report). The subsequent simulation study results do present, however, the sensitivity of crosswind hovering capability to the magnitude of the side force coefficient for an interference free configuration.

5.11.3.2.2.2 Dynamic Characteristics

For non-zero flight speeds the dynamic derivatives for the hull were developed from a combination of data and techniques from the Goodyear LTA Aerodynamics Handbook and Reference 11. The derivatives for the hover condition were estimated based on cross flow drag concepts.

Rotor forces resulting from angular velocities p, q, and r about the heavy lifter mass center are expected to be the principal helicopter contributions. In general, these forces depend upon a large number of variables, including rotor geometry, cyclic and collective pitch, rotor thrust coefficient, altitude, and relative velocity between rotor and free stream. To determine the forces for arbitrary values of these variables requires a level of simulation that was far beyond the scope of the current effort. The only reasonable alternative was to make use of existing helicopter data, which means that in general only those conditions for which data are available can be simulated.

Data for the CH-53D helicopter, which has the same dynamic system as the CH-54B, were obtained and examined. In essence, the data consisted of derivatives of three forces and three moments with respect to three linear and three angular velocities at various flight conditions. A major limitation of the data involved rotor thrust levels, since the heavy lifter is operable at thrust levels much less than those needed for flight with an isolated helicopter.

5.11.3.2.2.3 Gust Model

In order to secure a preliminary assessment of the responses of the HLA to a lateral gust encountered during the precision hovering mode of operation it was necessary to model the interaction of the gust front with the hull. The scope of the effort necessitated a somewhat simplistic approach to the problem which is regarded generally as unrealistically severe in terms of the rate at which the hull experiences the increasing gust load. Even in view of this, the vehicle response which is presented in Section 5.11.3.3 appears to be very acceptable.

Data from Reference 12 indicates that for cylinders impulsively started from rest, a steady state condition is reached within 2.5 to 4.0 cylinder diameters for laminar flow conditions. While the flow over the hull will generally be turbulent, this data has been used herein as representative of interaction distance of the gust front and hull. A first-order exponential indicial function has been considered with 93% of the gust velocity achieved after two hull diameters. As indicated in Section 5.11.3.3.2 sharp edge as well as sinusoidal gusts have been considered in the vehicle response studies during the precision hovering mode of operation.

Substantial additional effort is required in this area in order to secure actual results from which vehicle and control system design decisions can be made. These efforts would include better definition of the interaction of the gust with the hull at all angles of sideslip, a better definition of the applicable gust structure, and a model that accounts for the reaction of the gust front with the individual rotors.

5.11.3.2.2.4 Control Laws

The control laws define the manner in which the available control forces are mixed to secure the desired vehicle controllability. The control laws which have been developed are described in Section 6.9 and that description is not repeated here. The control laws developed to date should not be considered as final although they do appear to offer the needed vehicle response. The possibility of using the phenomenon of modification of the flow field around the hull by appropriately varying the individual rotor thrust levels has not been modeled and evaluated in the simulation. The possibility of doing this was revealed in the HLA exploratory wind tunnel testing (see Book III of this volume of the report). Additional testing is required to more completely understand this possibility and to obtain sufficient data to model it for simulation purposes.

5.11.3.2.2.5 HLA Autopilots

Most airships traditionally have been flown without autopilots or stability augmentation with the handling characteristics generally regarded as adequate. The autopilots provided on some of the larger airships provided pitch and heading hold for pilot relief.

An autopilot is required on the HLA for the precision hover mode and pitch attitude hold. The precision hover mode uses a position sensor that commands the autopilot which provides load spotting accuracy beyond the capabilities of a pilot manually flying the HLA via the primary flight control system in a hover mode. Another autopilot function which is important is pitch attitude hold with trim capability. This is important because the HLA does not have elevator surfaces to control the pitch attitude. Fore and aft motion of the control stick produces longitudinal thrust like a helicopter but the difference is that the pitch attitude of a helicopter does not produce much lift and is therefore relatively unimportant. The HLA develops a large force in the vertical direction with angle-of-attack (α) and it is therefore important that the angle-of-attack be controllable through the use of pitch attitude control. Since the fore-aft motion of the control stick does not control pitch attitude it must be controlled either by adding another pilot function or providing automatic control. The preferred solution is thought to be to provide an autopilot to maintain the HLA level with trim capability. The alternative of providing the pilot with manual differential control of the main rotors to control pitch is undesirable because the pilot's two hands are already occupied with the collective and cyclic sticks and it is not considered advisable to provide a third manual primary control. The static trim capability afforded by the fore and aft ballonets are integrated into the pitch attitude hold autopilot function.

Since hover and pitch attitude hold autopilots are required, it is natural to include heading hold for cruise flight since it will be desirable to have a considerable ferry capability with this vehicle. To totally control the HLA in these modes requires six autopilots for all six degrees of freedom. In this study five autopilots were implemented on the computer simulation to demonstrate the precision hover capability and heading control throughout the range of airspeeds. A block diagram of these autopilots is shown in Figure 5.16. The translation autopilots for hover control have position, rate and integral feedback for good control. Each block shows the autopilot gain where an output of 100 equals 100% of the control capability. For example in the longitudinal (X) direction an error of 0.61 m (2 feet) will provide the maximum control available disregarding the integral control. Integral control is provided in the hover translational and heading modes to drive the X, Y and ψ errors to zero for a steady wind condition. The rate feedback enhances the aerodynamic damping for good vehicle response. The attitude autopilots are the same for hover and cruise except yaw which has integral control in hover for more accurate control.

Pitch and roll attitude control is provided to maintain the HLA level for working close to the ground. In normal cruise flight the roll autopilot is not engaged since the metacentric moment provides adequate roll attitude stabilization.

The autopilot gains were determined experimentally using the HLA 6 DOF hybrid computer simulation to evaluate the performance over the range of airspeeds from hover to cruise velocity. The autopilot gains were found not to be very critical to provide good performance and no gain changes were required from zero to cruise velocity which indicates favorable flight characteristics for control.

These autopilot gains were not optimized and are therefore considered preliminary but indicative of the expected performance.

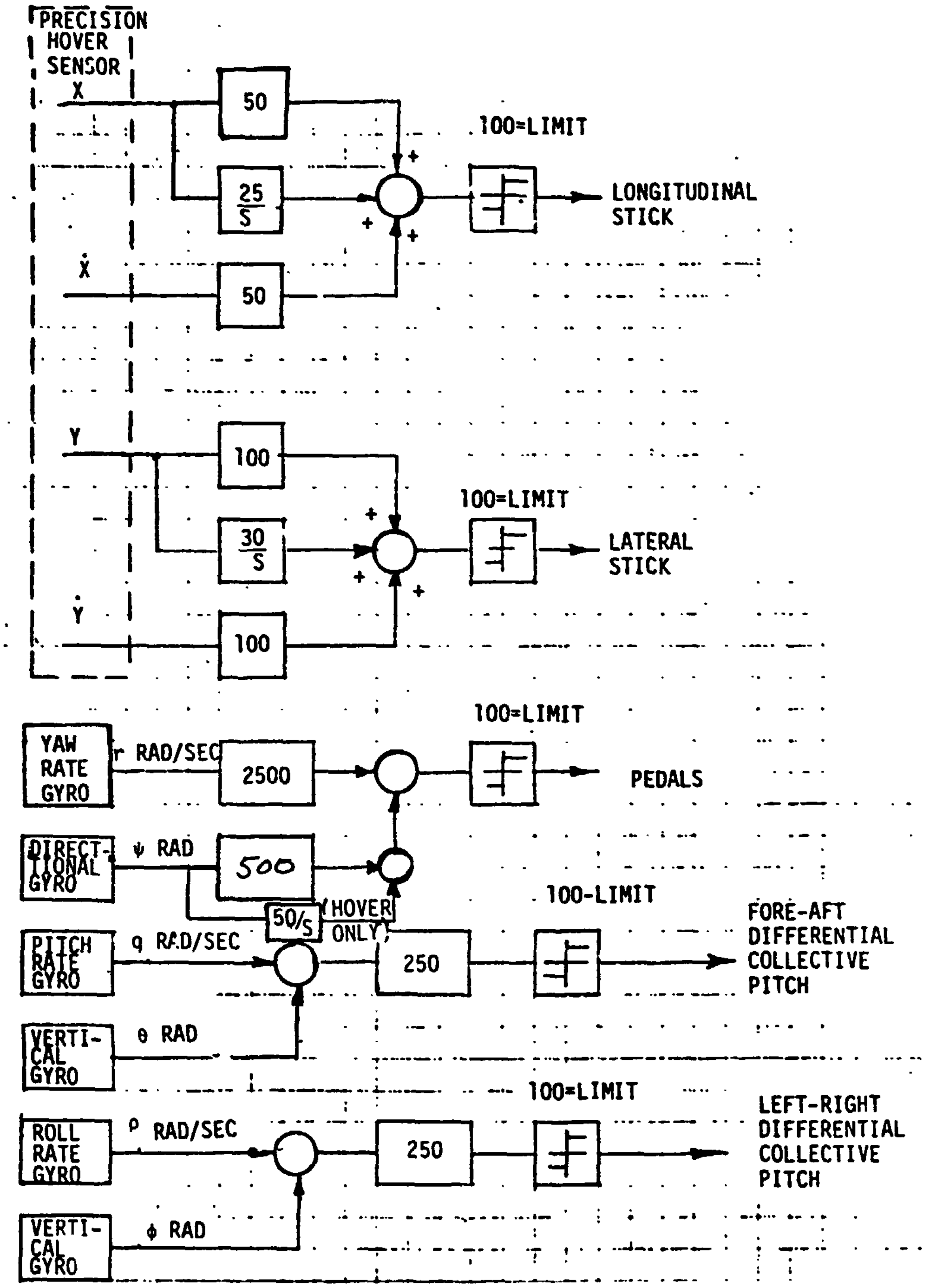


Figure 5.16 Autopilots - Longitudinal Axis

5.11.3.3 Results of 6 DOF Simulation Studies

5.11.3.3.1 Directional Stability

The availability of large propulsive forces permits the consideration of their use for inflight directional stability and accordingly the deletion of the empennage. Conventional airships have good directional stability characteristics at higher airspeeds where the dynamic pressure is sufficient to create large control forces (via rudder deflections). At lower airspeeds, the directional stability characteristics of the conventional airships become increasingly inadequate and eventually cannot adequately overcome the inertial forces. The use of propulsive thrust for directional control in the HLA reverses the trend of controllability with speed. Table 5.6 compares the available control moment to unstable aerodynamic moment ratio for the Phase II HLA and the ZPG-3W airship for various forward speeds. The larger this number at any speed the more favorable the directional controllability excluding inertial effects. The ZPG-3W was the last, and largest non-rigid airship built by Goodyear for the U. S. Navy. From the table it is seen that the directional controllability of the HLA at maximum gross weight is superior at all velocities to that of the ZPG-3W airship excluding the respective differences in vehicle yaw inertias. The yaw inertia of the HLA is greater, thus its directional stability with respect to the ZPG-3W for the above conditions is actually greater than indicated in the comparative numbers of Table 5.6.

At the lower airspeeds the HLA at maximum gross weight has far superior directional controllability as indicated in Table 5.6 which means the HLA eliminates the severe deficiency of past airships with respect to lack of low speed control.

The directional stability of the Phase II HLA at minimum gross weight can be enhanced to the same level as when flying at maximum gross weight by momentarily increasing main rotor collective pitch. This results in large main rotor differential cyclic pitch forces being available for directional control. This control law was modeled and included in the simulation on a trial

TABLE 5.6 COMPARISON OF DIRECTIONAL CONTROL CHARACTERISTICS FOR ZPG-3W AND PHASE II HLA

Velocity	ZPG-3W			Phase II HLA Configuration					
				Maximum GW			Minimum GW		
	Aerodynamic Moment ft-lbs/deg.	Available Control Moment ft-lbs	Ratio*	Aerodynamic Moment ft-lbs/deg.	Available Control Moment ft-lbs	Ratio*	Aerodynamic Moment ft-lbs/deg	Available Control Moment ft-lbs	Ratio*
35 kts	0.086×10^6	0.93×10^6	10.8	0.18×10^6	6×10^6	33.3	0.18×10^6	2×10^6	11.1
60 kts	0.25×10^6	2.7×10^6	10.8	0.5×10^6	6×10^6	12.0	0.5×10^6	2×10^6	4.0

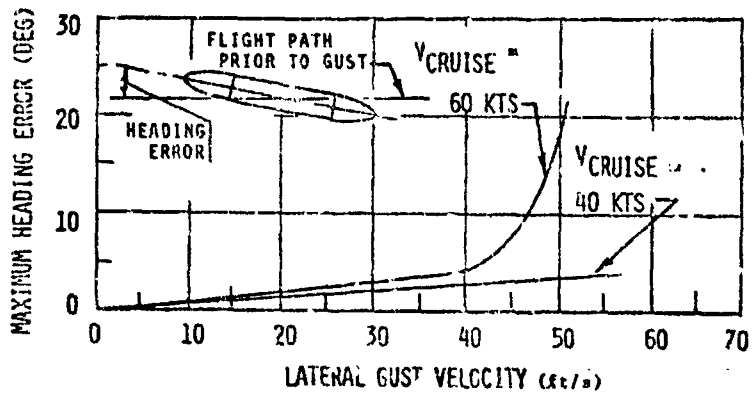
*Ratio of the available control moment to the aerodynamic moment

NOTE: 1.0 ft-lb = 1.382×10^{-1} m-kg, 1.0 kt = 5.14×10^{-1} m/s

basis and it was found that the altitude increase associated with this approach (due to a net vertical force) was less than 45.72 m (150 feet) even for an unrealisticably severe continuous sharp edge gust. As a result it appears this approach would be an acceptable procedure in many operations if required when flying at or near minimum gross weight.

In the Technology Assessment Analysis (Section 10.7.3) operational configurations are discussed that permit the main rotors to be used for directional control without the unwanted vertical forces. It is recommended in Section 10.1 that after appropriate analysis that promising advanced configurations be evaluated via actual modification to the initial flight research vehicle. As discussed in Section 10.7.3 the advanced configurations offer reduced operating cost as well as the type of performance benefit discussed here.

The directional control characteristics of the HLA as defined by the flight dynamics simulation are shown in Figure 5.17. A



NOTE: 1.0 kt = 5.144 x 10^{-1} m/s, 1.0 ft/s = 3.048 x 10^{-1} m/s

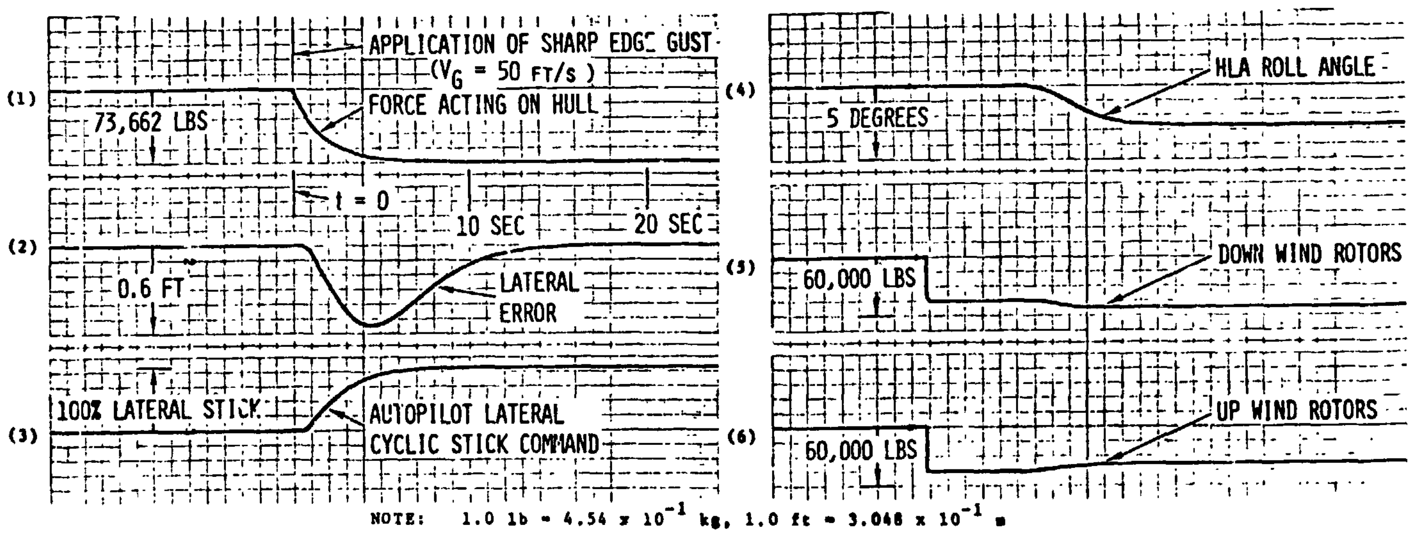
Figure 5.17 HLA Controllability Limits At Design Gross Weight With Current CH-54B Tail Rotor Power - 6 DOF Computer Simulation Results for a Continuous Sharp Edge Lateral Gust

continuous sharp edge gust is considered as the disturbance in Figure 5.17. This disturbance is simple to model but as noted earlier is in general also unrealistically severe in that it is considered to persist indefinitely. As illustrated in the figure, the autopilot is able to maintain good directional control for gusts up to 15.43 m/s (50 FPS) for the design forward speed of 30.86 m/s (60 knots). The maximum heading error is less for an original forward speed of 20.58 m/s (40 knots) which is consistent with ratios of Table 5.6. On the basis of the above discussion and simulation results, it is believed that the exclusive use of propulsive thrust for directional control is justified. Note that the simulation results of Figure 5.17 consider the tail rotor power currently available on the CH-54B. While increased power at the tail rotor would permit increased directional control, there does not appear to be any requirement for greater control in an initial vehicle.

5.11.3.3.2 Precision Hovering Studies

The precision hovering response of the HLA in the presence of a crosswind has been evaluated with the 6 DOF flight dynamics simulation. Prior to the power-on wind tunnel results being available, a study of the crosswind hovering characteristics of the HLA was made based on an interference-free configuration. Figure 5.18 illustrates typical simulation output parameters recorded during the precision hovering study. The particular case under study in Figure 5.18 is one which uses the side-force coefficient (without rotor effects) obtained during the recent HLA wind tunnel testing. As noted in Section 5.11.3.2.2.1, this coefficient may be larger than the full-scale value. Thus, the displacement pictured on Figure 5.18 may be larger and the crosswind velocity in which position can be held smaller than for an interference free full scale condition.

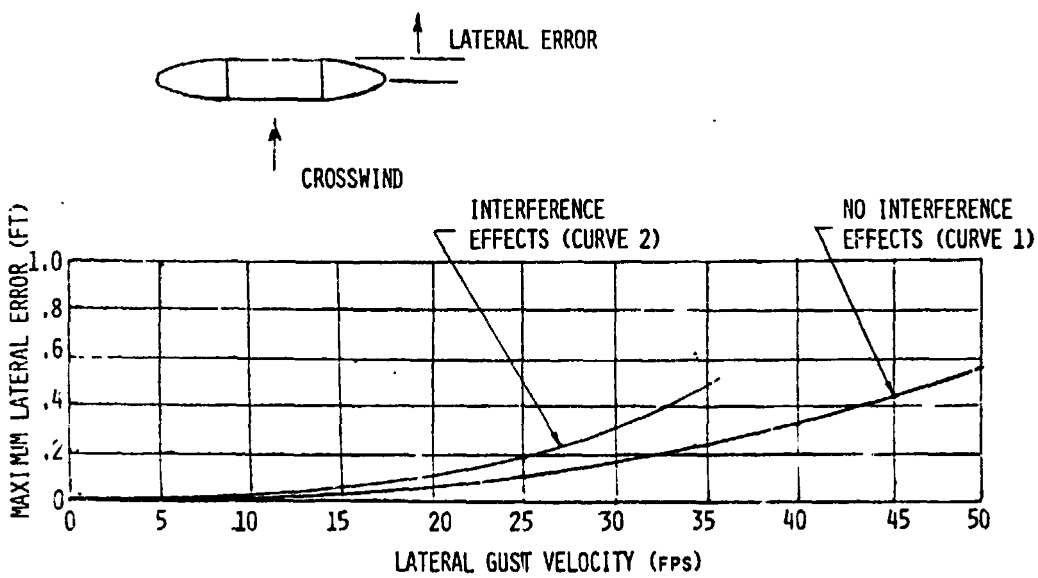
The first trace of Figure 5.18 indicates the point at which the gust is encountered and the subsequent increase in hull side force with time. Trace two depicts the lateral error as a function



NOTE: 1.0 lb = 4.54×10^{-1} kg, 1.0 ft = 3.048×10^{-1} m

Figure 5.18 Precision Hovering Response at Design Gross Weight - Six DOF Computer Simulation Results For A Continuous Sharp Edge Lateral Gust of time. The error, which reaches a maximum value of 0.17 m (0.55 feet) some 4.5 seconds after the gust front is encountered, is nulled by the autopilot within 15 seconds. The third curve shows the response of the cyclic sticks to the autopilot command. The fourth curve illustrates the roll angle resulting from the wind load applied at the hull center of buoyancy which is above the center of gravity of the vehicle. As a result, the HLA tends to roll but is restored to the degree shown by the metacentric moment and the roll autopilot (which calls for differential collective pitch that results in the propulsive restoring moment). The fifth and sixth traces indicate the downwind and upwind rotor thrust levels and variations in response to the autopilot commands required to maintain trim.

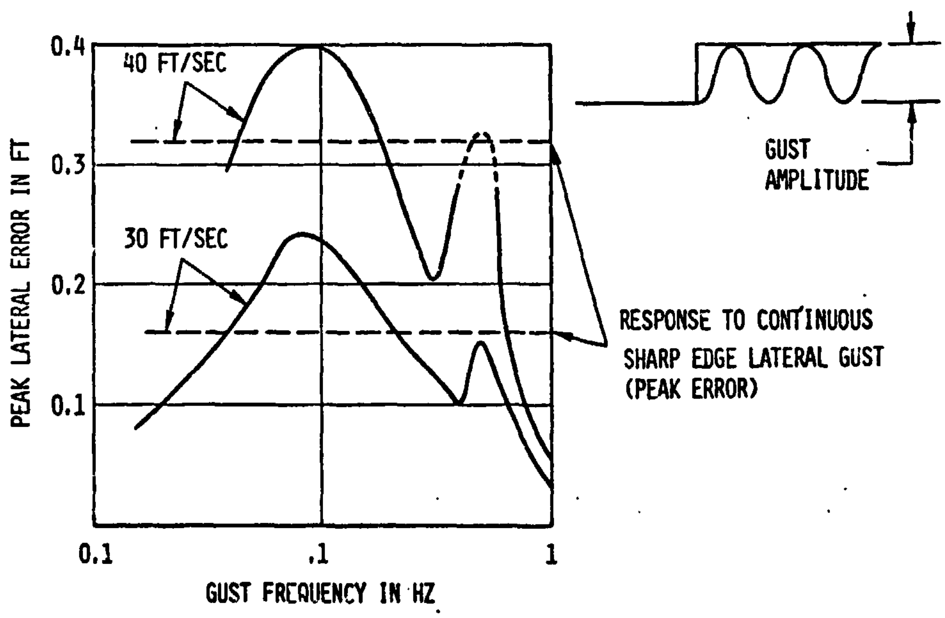
Curve one of Figure 5.19 presents the maximum lateral error for lateral gust velocities up to 15.24 m/s (50 fps) for the side force coefficient used in Figure 5.18 (i.e., the HLA wind tunnel data without rotor effects). Curve two illustrates the adverse effect for the rotor induced interference observed during the



NOTE: 1.0 ft = 3.048 x 10^{-1} m, 1.0 ft/s = 3.048 x 10^{-1} m/s

Figure 5.19 Peak Lateral Precision Hovering Response at Design Gross Weight - Six DOF Computer Simulation Results for a Continuous Sharp Edge Lateral Gust wind tunnel testing. The crosswind hover interference model has been included in the 6 DOF simulation to obtain the results of curve two of Figure 5.19. As indicated the interference effects have a considerable adverse effect on cross wind hovering capability. The approaches for improving the crosswind hovering capability of the current configuration as discussed in Section 5.11.2 will, of course, tend to translate curve two toward curve one as well as increase the magnitude of the crosswind in which station can be maintained.

A continuous sinusoidal lateral gust was used to determine the frequency response of the precision hovering system autopilot loop (see Figure 5.20). Again, this represents an overly severe condition. The frequency response characteristics show that the loop is rather well damped. The sharp edge gust response is repeated from Figure 5.19 for comparative purposes. The response to the sinusoidal input is about 3 db greater than the peak error from a step response.



NOTE: 1.0 ft = 3.048×10^{-1} m, 1.0 ft/s = 3.048×10^{-1} m/s

Figure 5.20 Frequency Response at Design Gross Weight For A Continuous Sinusoidal Gust

5.12 Payload Suspension

The simulation results of the prior sections are based upon a payload rigidly attached to the HLA. For some payload attachment arrangements, this may be an acceptable approximation in terms of analyzing the vehicle flight dynamics. In general, however, it is not sufficiently representative and as recommended in Section 10, future effort in the flight dynamics area must involve modeling of the payload dynamics. The payload dynamics must also be modeled because the response of the hull to turbulence will result in payload dynamics that must be accounted for in evaluating cargo placement capabilities.

A few comments are in order relative to this subject, however, in order to put into proper perspective the subject of payload dynamics of the HLA relative to that of existing helicopters. It should be noted that the maximum payload to vehicle empty weight ratio for the HLA is approximately 1.0 whereas for the crane helicopter it is more like 1.25. These ratios alone suggest, from the standpoint of the payload affecting the flight dynamics of the vehicle, that the HLA should be less sensitive than the crane helicopter. The HLA also has large apparent masses which would further mitigate payload induced dynamics.

In general, existing helicopter experience, in terms of cargo suspension or sling techniques, will serve as meaningful background for the HLA. The existing experience requires extension, obviously, in terms of load capability of the cargo handling system as discussed in Section 10.

5.13 Center Point Mooring Concept

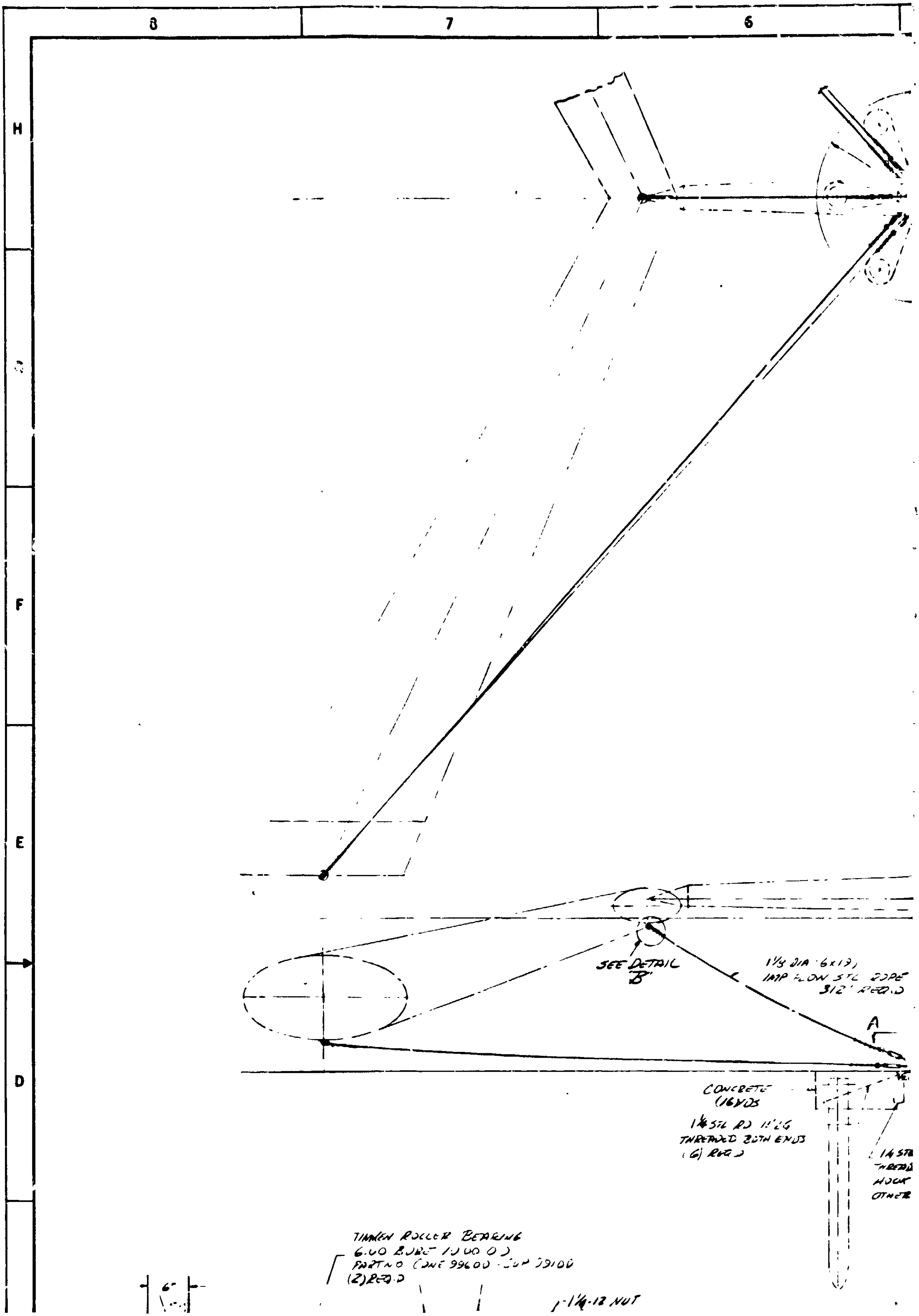
Section 5.3.2 presents the basic dilemma that exists relative to including or excluding a vertical tail on the HLA. Based upon the data of References 9 and 10 it was initially known that a tail, because of the associated increase in side-force coefficient at large angles of sideslip would appreciably reduce the magnitude of the crosswind in which the HLA could maintain station. The wind tunnel tests (see Book III of this volume of the report) have revealed for the current configuration that the rotors induce flow over the hull which effectively increase its sideforce coefficient in a crosswind condition. Accordingly, it can be surmised that with a tail sufficiently large to permit consideration of conventional bow mooring, the increase in sideforce coefficient would be even larger, due to the increased vertical area, than the data of References 9 and 10 originally indicated. Based on this factor and the results of the flight dynamics simulation studies presented in Section 5.11, which indicate that adequate directional stability can be obtained without a vertical tail, the Phase II configuration excludes a vertical tail.

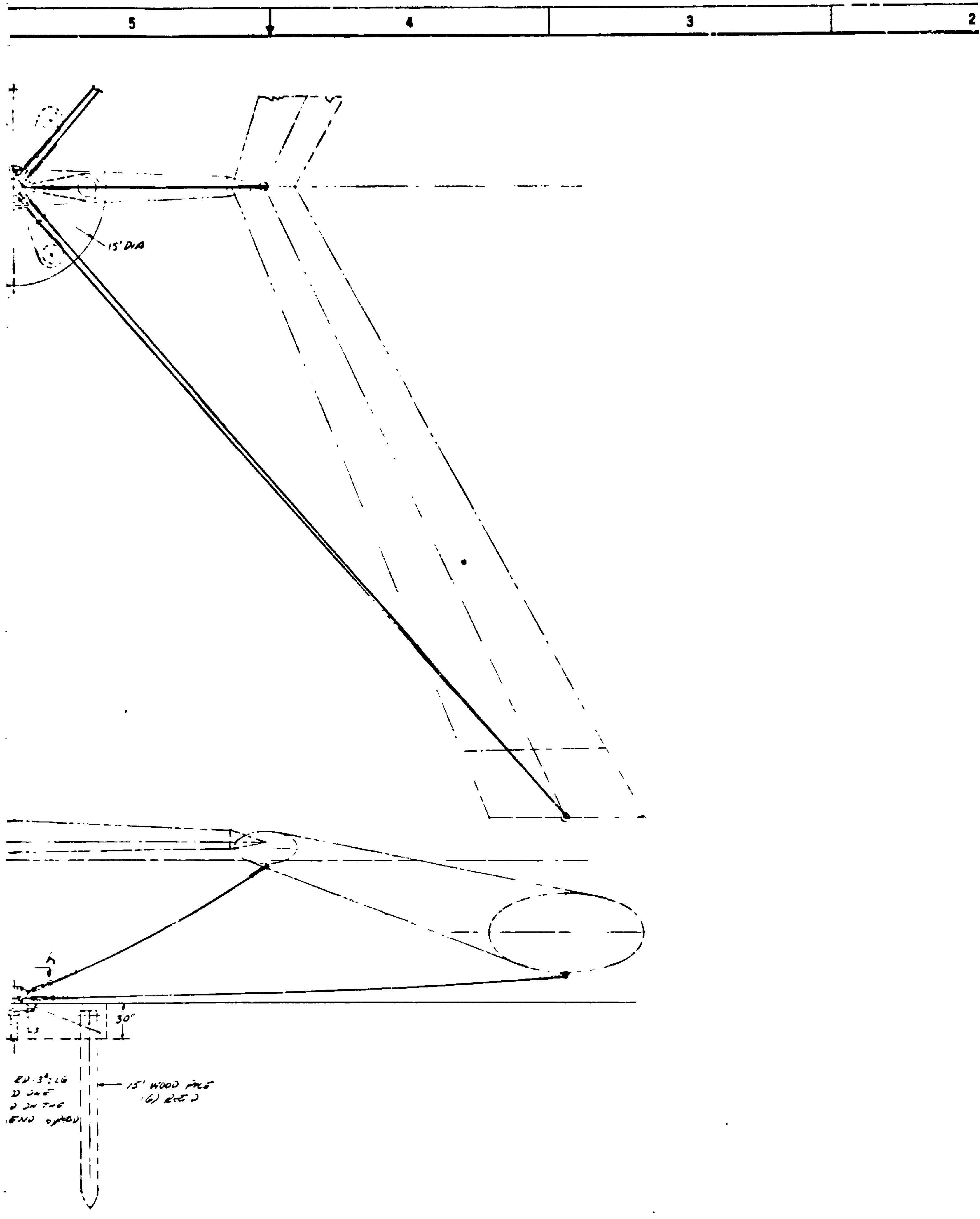
The absence of a tail coupled with a desire to improve past airship mooring techniques led to consideration of what has been termed the center point mooring concept. The aerodynamics of the HLA without a tail (see Book III of this volume of the report) creates a stable condition with the hull broadside to the wind ($\beta = 90°$) when moored to a pivot located at the ground plane at the center of the planform. As indicated in Figure 5.21 (GAC Drawing 76-333) cables connect the pivot to frame joints F1, A1, LF1, RF1, LA1 and RA1.

The most severe distortion of the envelope cross section occurs in the center point mooring condition. The current analysis is based on pressure distribution data in Reference 8 and available statistical data relative to wind magnitudes and frequencies. Based on this data, a 65 mph broadside wind condition was selected for analysis. The above pressure distribution does not include ground plane effects and it is known from References 9 and 10 that considerable effects from the ground plane will occur. Exactly how the ground plane will affect the pressure distribution over the hull and the resulting fabric loads is not known at this point. As a result of the general lack of aerodynamic data at large angles of sideslip, the recommended FRV development program includes sufficient wind tunnel testing to obtain the data needed to support the FRV development. This additional testing requirement is not only required in order to finalize the mooring concept

FIGURE 5.21 CENTER POINT MOORING DETAILS
(GOODYEAR DRAWING NO. 76-333)

FOLDOUT FRAME 1

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15' DIA
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FOLDOUT FRAME 3

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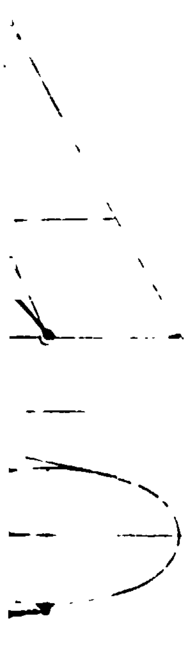

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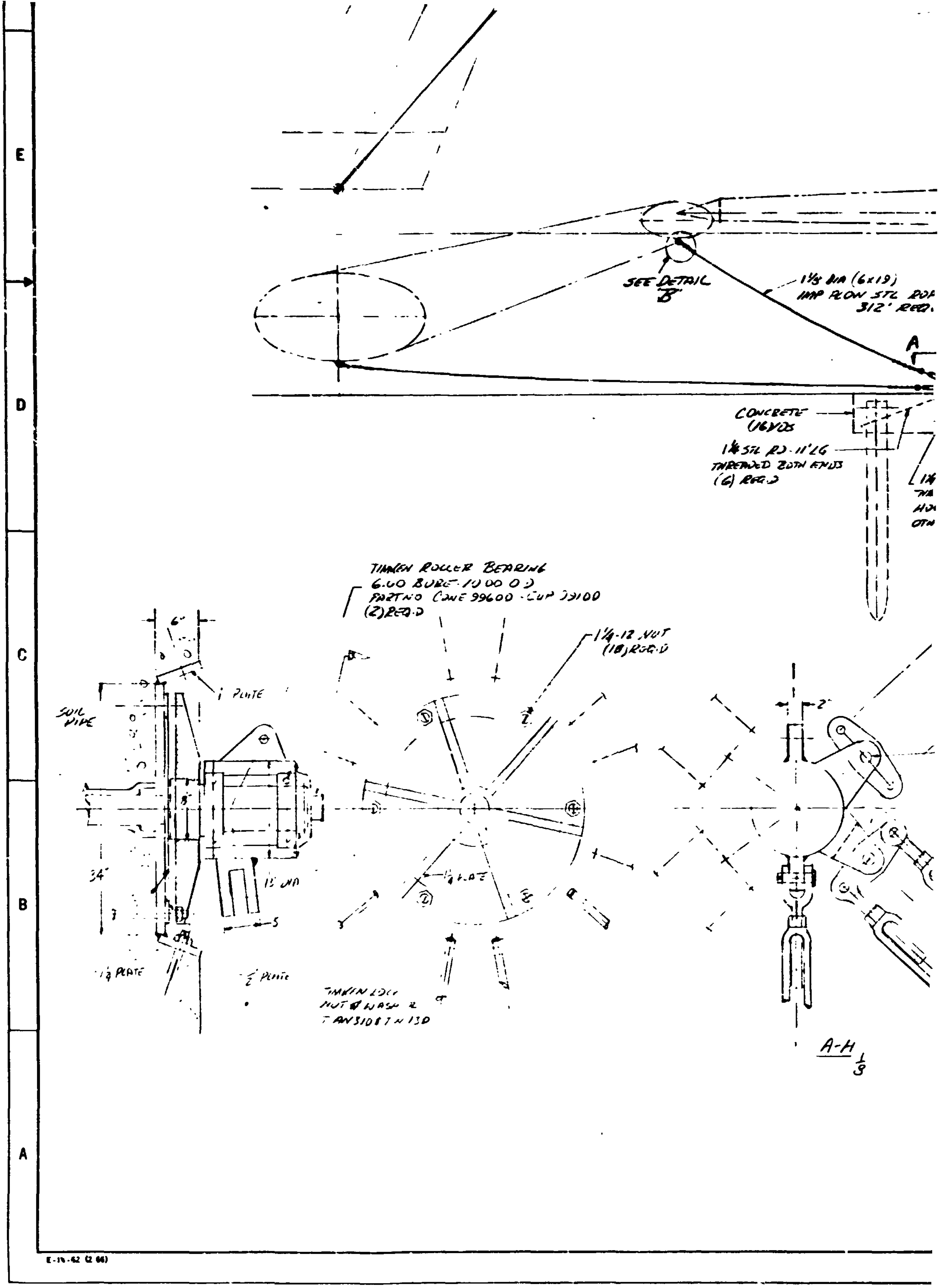

FOLDOUT FRAME 4

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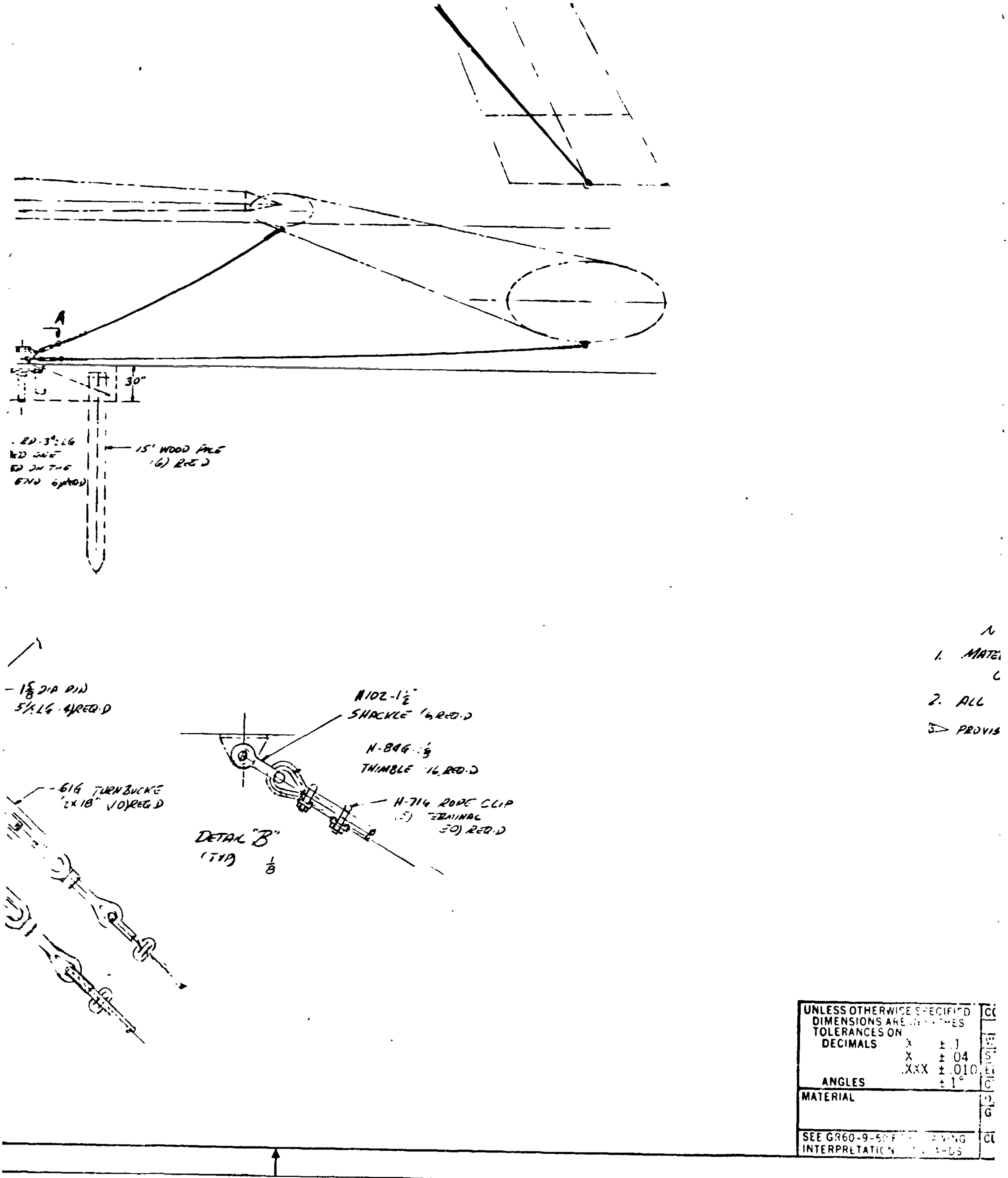

FOLDOUT FRAME 5

NOTE:

1. MATERIAL MAY BE PURCHASED FROM ...SON-HAMILTON CO CLEVELAND OHIO - SUBSIDIARY OF ANVIL IND.
2. ALL PLATE 1020 OR EQUIV LOW CARBON STEEL
3. PROVISIONS FOR SHORE POWER JUNCTION BOX

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FOLDOUT FRAME 6

but is also essential in the overall design of the HLA as discussed subsequently. As in many areas, however, the FRV is needed for final mooring system definition.

If the additional wind tunnel data indicate that the required fabric strengths are sufficiently high that the resulting weight penalties offset the advantages offered by the center point mooring concept a more conventional mooring concept would be adopted.

6.0 HLA FLIGHT CONTROL SYSTEM

6.1 General

Control of the HLA is accomplished through the appropriate combination of available rotor forces thus conventional airship aerodynamic control surfaces are not required. As a result of this control approach a severe deficiency of past airships (i.e. lack of low speed control) is eliminated since airspeed is not required for the HLA to develop maximum control forces.

The HLA is flown using standard helicopter controls. The aft left helicopter serves as the command station in which a command and safety pilot are located. The command pilot's conventional mechanical controls are replaced with electric cyclic and collective sticks as well as electric pedals which generate the fly-by-wire (FBW) commands to control all four helicopters. A simplified block diagram of the FBW control system is shown in Figure 6.1. The control system includes an automatic flight control system (AFCS) and a precision hover sensor (PHS) for automatic precision hover control. These components were developed in the HLH program.

All the aerostat manometer indicators (air and helium) as well as the ballonet blower, damper, and air and helium valve controls will be located in the command helicopter. An automatic envelope pressure control system is utilized with a manual backup system also provided.

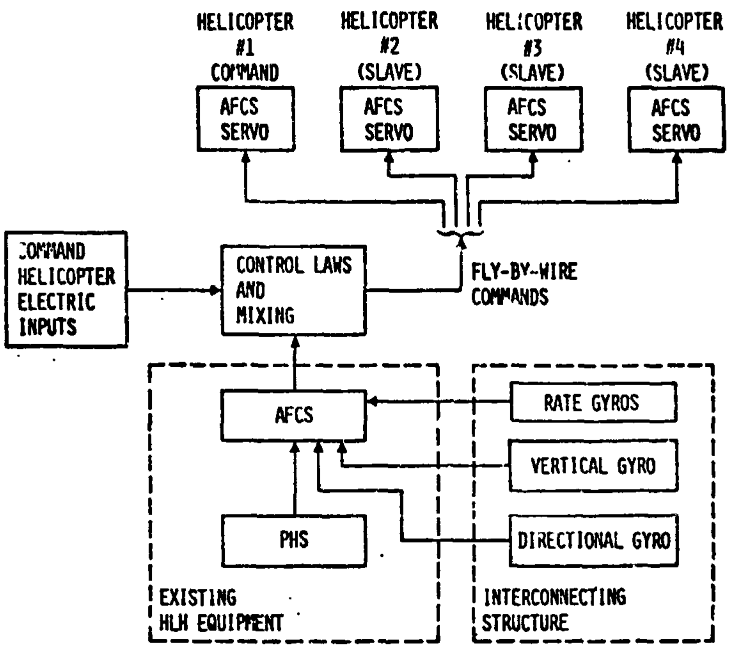


Figure 6.1 HLA Fly-By-Wire Control System Block Diagram

6.2 Control Force Configuration

The main and tail rotors of the four helicopters result in eight individual forces with which to control the HLA. The CH-54B tail rotors are replaced with variable pitch propellers to increase the thrust over the HLA speed range. The tail propellers on the two aft helicopters are re-oriented 90° to provide thrust along the helicopter's longitudinal axis. The helicopters are attached to the interconnecting structure by a two axis gimbal which allows the helicopters to pitch and roll 15° relative to the HLA. This provides a means of vectoring the main rotor thrust into the HLA horizontal plane to augment the control forces from the tail rotors. By controlling the magnitude and direction of the eight available control forces, through a set of control laws, the HLA can be controlled in the six degrees of freedom.

6.3 Control of Helicopters on Gimbals

Initially a concept was considered in which the helicopters were rigidly attached to the interconnecting structure and as in the case of a helicopter, the main rotor thrust was vectored by use of cyclic pitch. However, this results in high bending loads in the main rotor shaft since the shaft is not free to realign with the tilted thrust vector.

Thus, it was necessary to attach the helicopters to the interconnecting structure with a two-axis gimbal. This allows the main rotor shaft to tilt with the thrust vector eliminating the excessive bending load.

Up to 4 degrees of cyclic pitch can be used on a steady state basis without reducing the main rotor shaft life. This can be used to increase the control forces in the horizontal plane by 25% when the helicopter has reached its gimbal travel limit of 15% resulting in an effective gimbal travel of $\pm 19°$.

The gimbal system consists of an outer (roll) gimbal and an inner (pitch) gimbal. The helicopter is controlled on the pitch gimbal by use of the main rotor longitudinal cyclic pitch which produces a moment about this axis. A servo control loop, using the helicopter pitch attitude autopilot, controls the helicopter on the pitch gimbal. The attitude command input is from the HLA fly-by-wire control system.

The main rotor reaction torque is counterbalanced by the tail rotor in a helicopter. However, in the case of the HLA configuration this torque is transmitted to the HLA. This torque results in a clockwise yawing moment which is counteracted by a yawing moment bias in the FBW control system. This is achieved by tilting the helicopters on the right side slightly forward and slightly aft on the left side.

As each helicopter is pitched on the gimbal from zero degrees, to achieve forward flight, etc., part of the main rotor reaction torque is coupled into the roll gimbal. This torque is counteracted by a pair of hydraulic actuators which are also used to provide the rotation of the helicopter on the roll gimbal.

6.4 HLA Flight Controls and Modes

The HLA is very similar to a conventional helicopter with respect to the required pilot functions. To take-off, the command pilot increases collective pitch which commands the four main rotors in unison and the HLA ascends vertically. Figure 6.2 shows how the thrust vectors are utilized to achieve control in six degrees of freedom. By pushing the cyclic stick forward the helicopters pitch forward on their gimbals developing a forward thrust which results in the desired forward velocity for the HLA. By pushing the cyclic stick in the lateral direction the helicopters roll in the same direction and the HLA translates laterally. The pedals control yaw. To yaw to the right the helicopters on the left side are pitched forward and the helicopters on the right side are pitched aft producing a yawing moment. The collective pitch of the tail propellers are synchronized with the cyclic stick and pedal commands to augment the main rotors and increase the thrust and moment effectiveness.

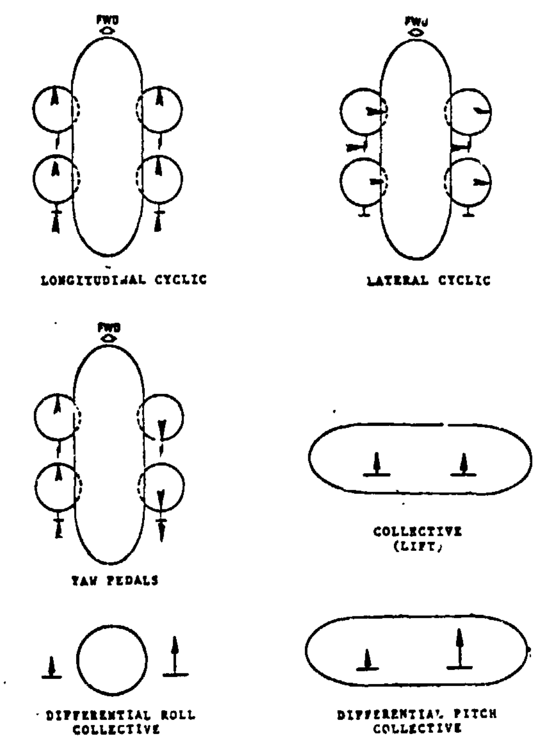


Figure 6.2 Thrust and Control Force Configurations

The only functions that are not manually controlled are the HLA roll and pitch attitude. The pitch attitude hold function with trim capability is an autopilot task which maintains the HLA in a level pitch attitude by differential collective pitch of the main rotors. This relieves the pilot of an additional control function. During normal flight the HLA metacentric moment in roll is sufficient to provide stability about this axis without external control forces. Roll attitude hold can be engaged by the pilot for precise control for hover and landing.

The command pilot with these capabilities can perform the basic flight modes of:

(1) Takeoff and landing

(2) Translation flight (longitudinal and lateral)

(3) Hover

All of these modes can be performed by the command pilot except when precise hovering is required. Typical of this situation is the extraction of containers from the hold of a ship or the placement of equipment or prefabricated structures on an existing foundation under gusty conditions.* To perform this precision hovering maneuver requires a precision hover sensor (PHS) and an automatic flight control system (AFCS) for the HLA to interface with the primary FBW control system. It was then natural to include the other standard automatic flight control modes of, heading hold, altitude hold and pitch attitude hold for ferry flight. As noted previously, the PHS and the AFCS hardware developed for the HLH have been adapted to the Phase II HLA.

The PHS (which measures X, Y and Z distances to an accuracy better than 1 inch) was developed by RCA and successfully flight-tested for over three hundred flight hours in a Boeing Vertol CH-47 along with the remaining HLH FBW control system elements. The precise location for the PHS on the interconnecting structure

*Because of this type of requirement an automatic precision hover control system was developed for the heavy lift helicopter which has the capability to control the helicopter very accurately for picking up and depositing loads in gusty weather.

has not been selected and this selection process will require considerable additional analysis. The PHS must be able to view the hover target unobstructed by the payload and must also be located in an area isolated from large aeroelastic effects. In addition, the PHS will require protection from the environmental elements.

6.5 Existing Flight Controls on CH-54B Helicopter

This section describes the CH-54B primary and automatic flight control system and a method for adapting the control system to interface with the HLA FBW concept.

The pilot and co-pilot have conventional helicopter controls and instruments. In addition, an aft pilot facing aft operates the crane. He is provided with an electric hover stick (cyclic and yaw) and a mechanical collective stick. The basic flight control system block diagram is shown in Figure 6.3. It is of great interest regarding the fly-by-wire capability that all the

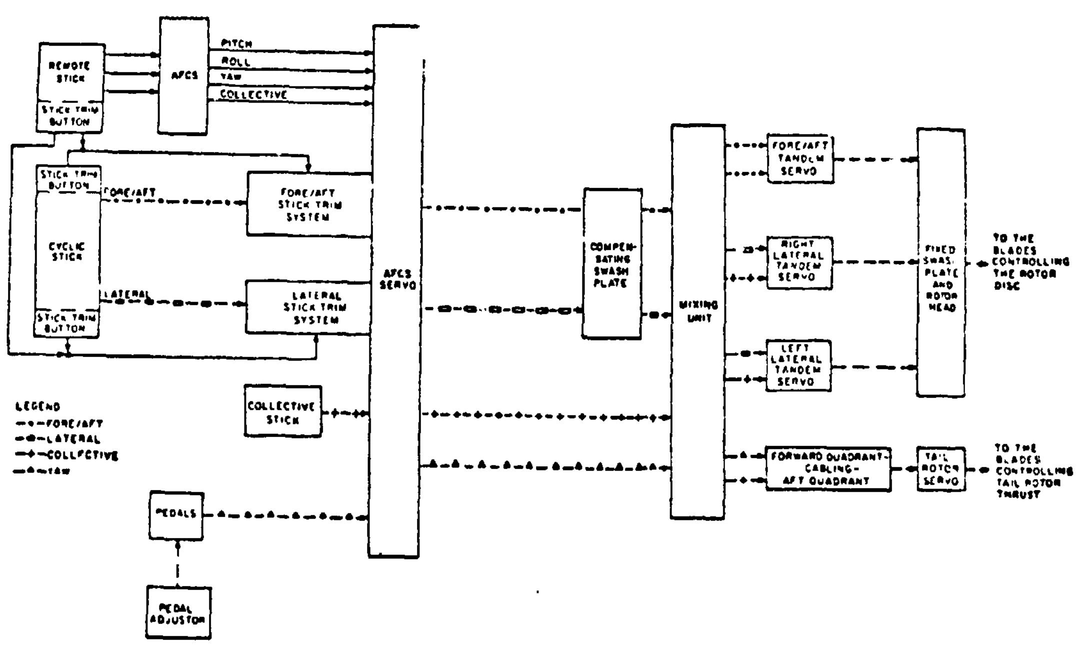


Figure 6.3 CH-54B Flight Control System Block Diagram

primary controls and the remote stick operate through the automatic flight control system servo (AFCS servo). The AFCS servo is the power boost in the mechanical flight control system which provides for electrical inputs with mechanical stick overrride capability by the pilot and co-pilot of any electrical signal which originates in either the stability augmentation system (SAS) or the autopilot (AP). The SAS and AP systems have limited authority of ±5% in pitch and yaw and ±10% in roll. However, there is another function, stick trim, which has full cyclic authority but at a limited rate of 12% of full travel per second. A simplified block diagram is shown in Figure 6.4 to further clarify the operation of the AFCS. This shows one axis of the cyclic stick control loop.

The AFCS servo sums the pilot's stick or autopilot command through the trim actuator with the stability augmentation signals into a control command to the mixing unit. The main rotor flight control schematic is shown in Figure 6.5. This illustrates how the primary flight control system operates mechanically through

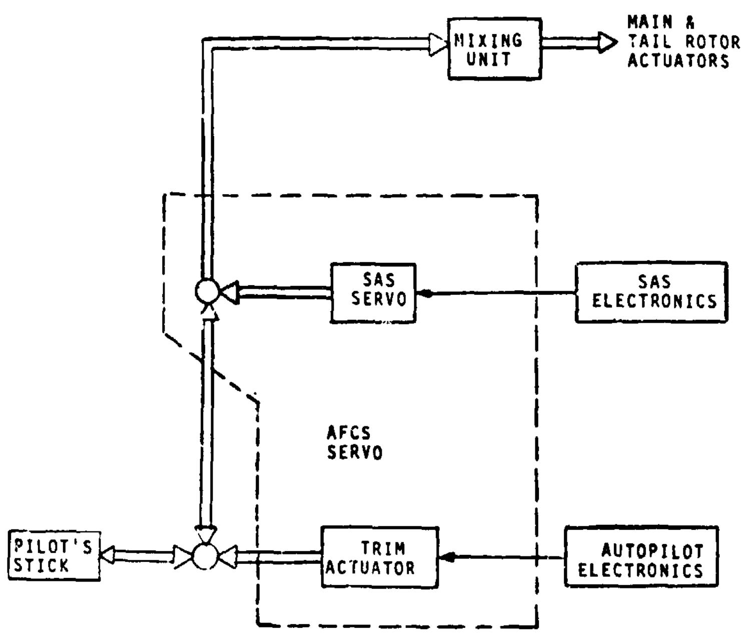


Figure 6.4 CH-54B Flight Control System

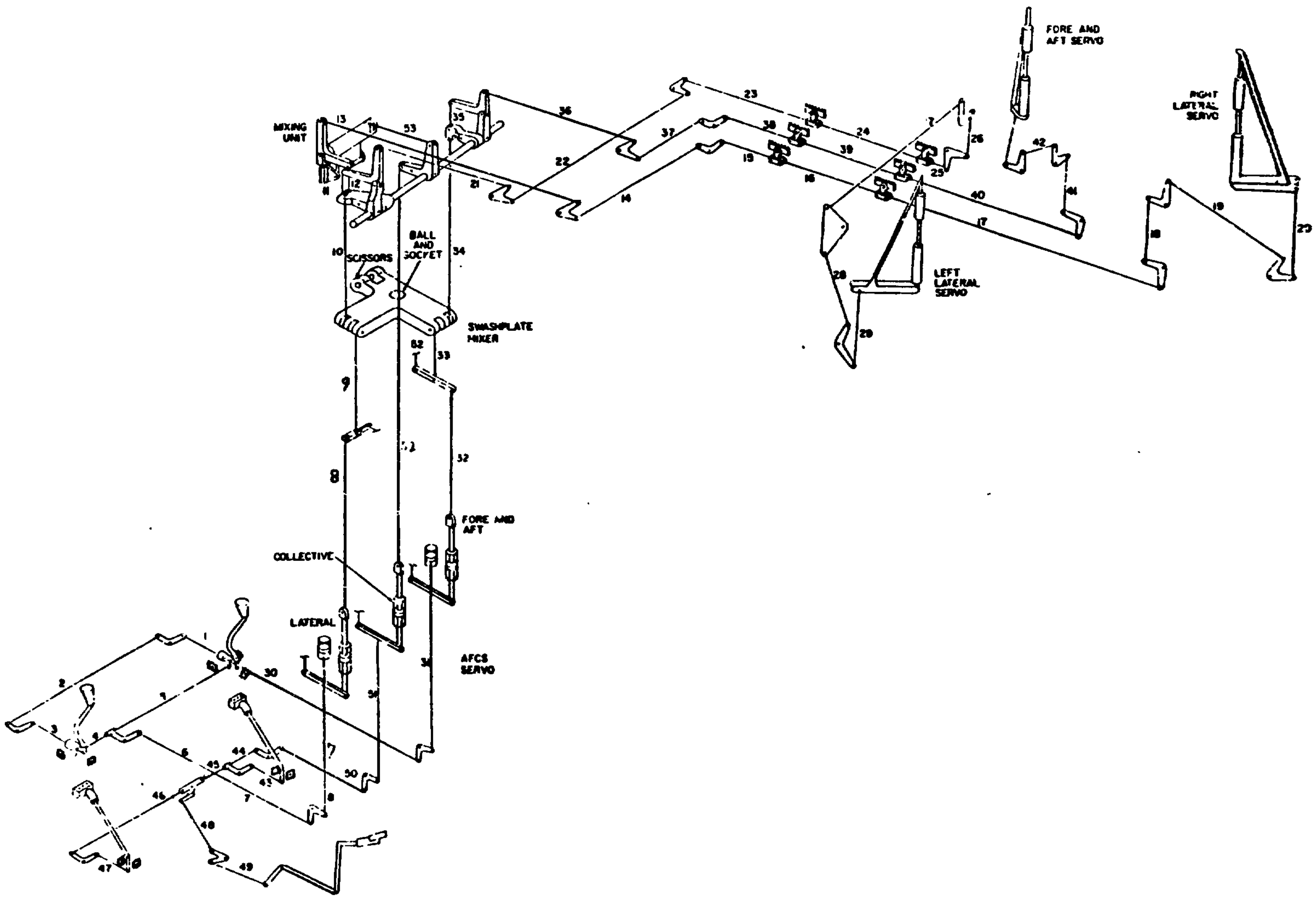


Figure 6.5 Main Rotor Flight Controls Schematic

the AFCS servos whether they are on or off. The collective stick AFCS servo has no trim actuator but it does have electrical input capability for the autopilot altitude hold mode.

6.6 Prior CH-54B Fly-By-Wire Conversion by Sikorsky

The CH-54B is ideally suited to conversion to fly-by-wire control with pilot override because of the AFCS servo. A CH-54B has been converted to fly-by-wire by Sikorsky for NASA Langley for variable stability test purposes. A block diagram of the pitch axis for that fly-by-wire system is shown in Figure 6.6. The evaluation pilot's cyclic stick was disconnected mechanically from the co-pilot stick and electrical transducers with a stick feel system were added to the (fly-by-wire) stick. The electrical stick position is summed electrically with pitch attitude and attitude rate feedback and drives the AFCS trim servo valve. The trim actuator drives the safety stick mechanically in the normal configuration with a limited rate of 12% of full travel per second although it has full authority. The trim servo valve

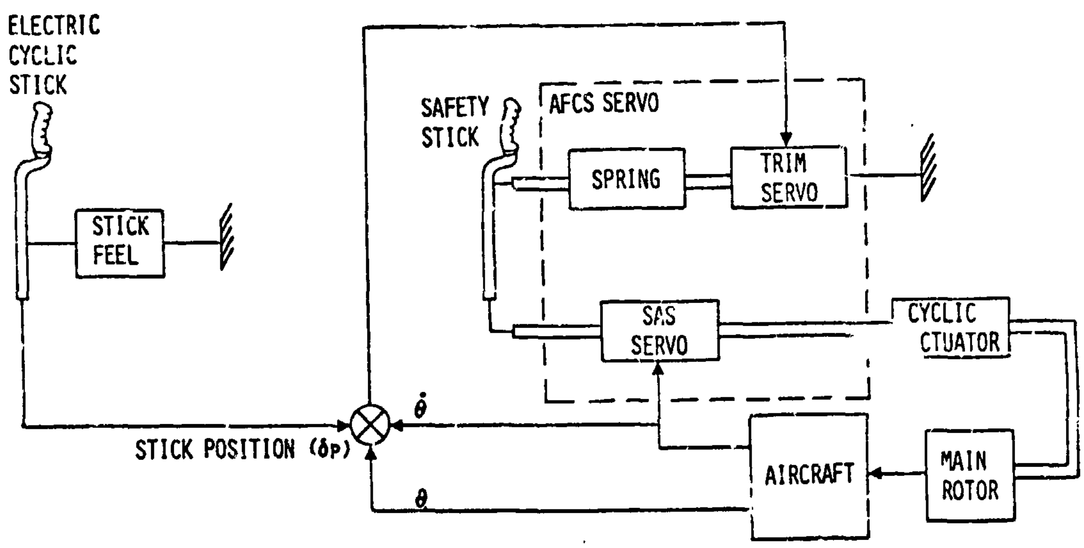


Figure 6.6 Sikorsky CH-54B Fly-By-Wire System Pitch Axis

was changed for this configuration increasing its rate limit to 100% of full travel per second to permit normal flight control input rates to be inserted electrically into the AFCS. The original inner stability augmentation loop on the AFCS was maintained.

The safety pilot has the ability to override the evaluation pilot at any time with mechanical stick positions taking command of the helicopter if requried. The roll and yaw axes were also implemented for fly-by-wire in this system but pilot's collective remained a mechanical input.

This system is not true fly-by-wire because the control system is primarily mechanical including mixing out to the mechanical actuation of the main and tail rotor servo actuators, but the ability to insert electrical stick signals into the AFCS servo to control the helicopter main and tail rotors with full authority has been demonstrated with safety pilot override. This is exactly what is required for the heavy lift airship concept where the primary inputs will be electrical from a command helicopter with the capability of the safety pilot to override the fly-by-wire input if required.

6.7 CH-54B FBW Requirements for HLA

The helicopters are attached to the HLA with roll and pitch gimbals which have a freedom of ±15° in each axis. Translational thrust augmenting the tail rotor thrust is achieved by tilting the helicopters about these axes to tilt the main rotor plane as commanded by the FBW inputs to the four CH-54Bs. There are four flight control commands to each helicopter:

(1) Main rotor collective pitch

(2) Tail rotor collective pitch

(3) Helicopter pitch attitude

(4) Helicopter roll attitude

The collective pitch signals can be inserted directly into the CH-54B AFCS servo which is modified to provide the full authority and adequate rate of collective application.

The FBW roll and pitch commands are inputs to the helicopter attitude autopilot which uses the helicopter's vertical gyro as a reference. As a result, the helicopter attitude commands are with respect to local vertical. The force to rotate the helicopter in pitch is longitudinal cyclic pitch of the main rotor which produces a torque to rotate the helicopter to the commanded pitch attitude. Servo controlled hydraulic actuators are used to tilt the helicopters in roll. Cyclic pitch cannot be used for roll because of the cross coupling of the main rotor shaft torque into this axis when the pitch gimbal attitude is not zero. If this torque were counteracted by lateral cyclic pitch of the main rotor the resulting lateral thrust would produce unwanted motion in the lateral direction.

6.8 CH-54B AFCS Servo Modifications for HLA

The key element to accomplish fly-by-wire is the attitude autopilot and the AFCS servo. The AFCS servo consists of four independent mechanical electro-hydraulic servos to control main rotor collective, lateral cyclic, longitudinal cyclic and tail rotor collective with mechanical or electrical inputs. The modifications required to this unit will be similar to the changes required for the Sikorsky FBW conversion. The FBW pitch input to the helicopter will be a pitch attitude command to the autopilot. The force to obtain this rotation is achieved by longitudinal cyclic pitch of the main rotor. Figure 6.7 shows the cyclic AFCS servo which will be used in the pitch axis. By removing the present stick trim servo valve which has a limited rate of 12% of full stick travel per second and replacing it with a valve with 100% travel per second capability as in the Sikorsky FBW conversion, the remote command pilot has good rate control with full authority. The other servo valve provides for an electrical input for inner loop stability augmentation. It will probably remain unchanged with limited authority but the electronics

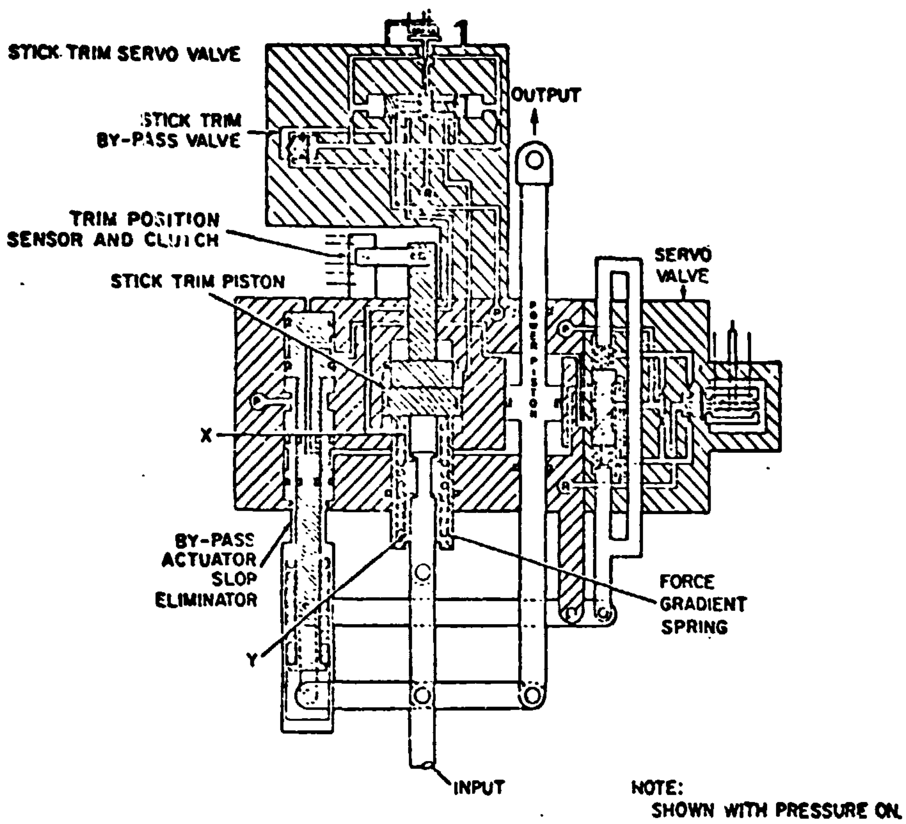


Figure 6.7 CH-54B - Cyclic AFCS Servo Schematic

involved in the feedback may be quite different because of the different dynamic characteristics of the gimbaled helicopter. The safety pilot can override the electrical stick input by the mechanical cyclic stick which is shown as an input at the bottom center of Figure 6.7. The safety pilot's mechanical stick is always connected to the AFCS and it will follow these motions. A stick force of ten pounds is required to overcome the electrical inputs.

The FBW roll input to each helicopter will be a roll attitude command. The force to obtain this rotation is achieved by hydraulic servo actuators which rotate the helicopter on the gimbal. Lateral cyclic pitch is not used so the roll autopilot will be modified to control the servoactuator instead of lateral cyclic pitch. This modification is more extensive than in pitch principally because of the addition of the actuator drive.

The collective AFCS servo schematic is shown on Figure 6.8. Unlike the cyclic servo it does not have a separate trim input. The electrical input to the servo valve has full authority at 100% per second rate with mechanical feedback to the collective stick. This electrical input would be used for the fly-by-wire control. Any modifications to the collective AFCS servo to meet the HLA requirements would be minor. It may be necessary to change the open loop spring to one with a higher spring rate to permit electrical inputs of higher frequencies. Further control systems analysis will be necessary to define these detailed requirements.

The yaw AFCS servo schematic shown on Figure 6.9 is very similar to the cyclic servo. The yaw trim servovalve will be replaced to extend the rate of authority for good control. The pilot's yaw pedals will be mechanically disconnected and electrical transducers added for fly-by-wire capability. The safety pilot yaw pedals will still be mechanically connected to the AFCS input. The yaw output shown on the tail rotors controls

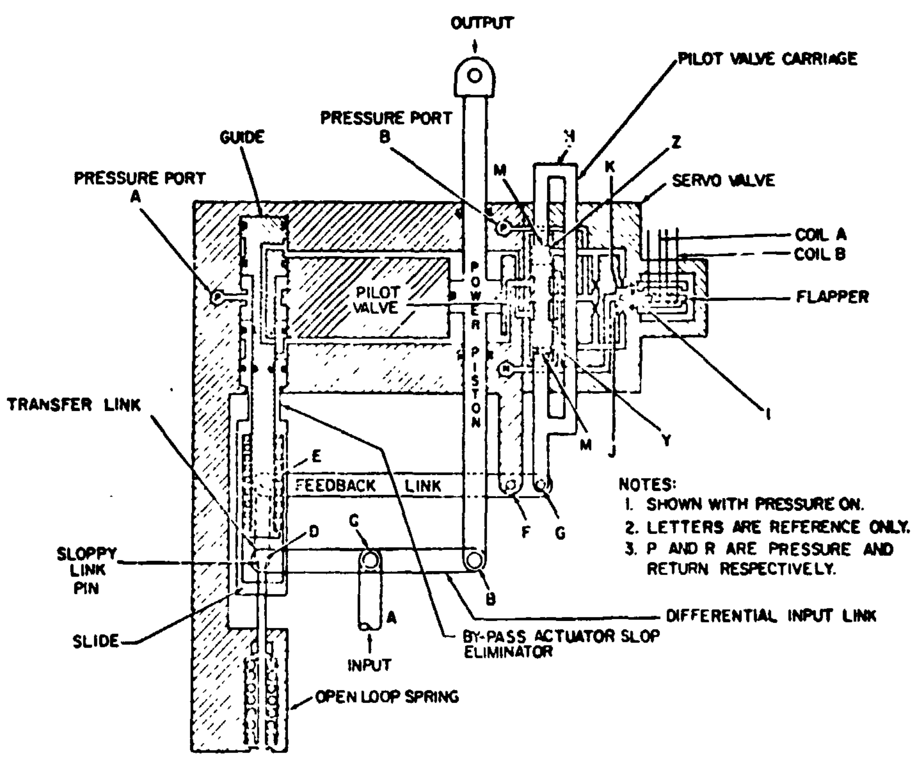


Figure 6.8 Collective AFCS Servo Schematic

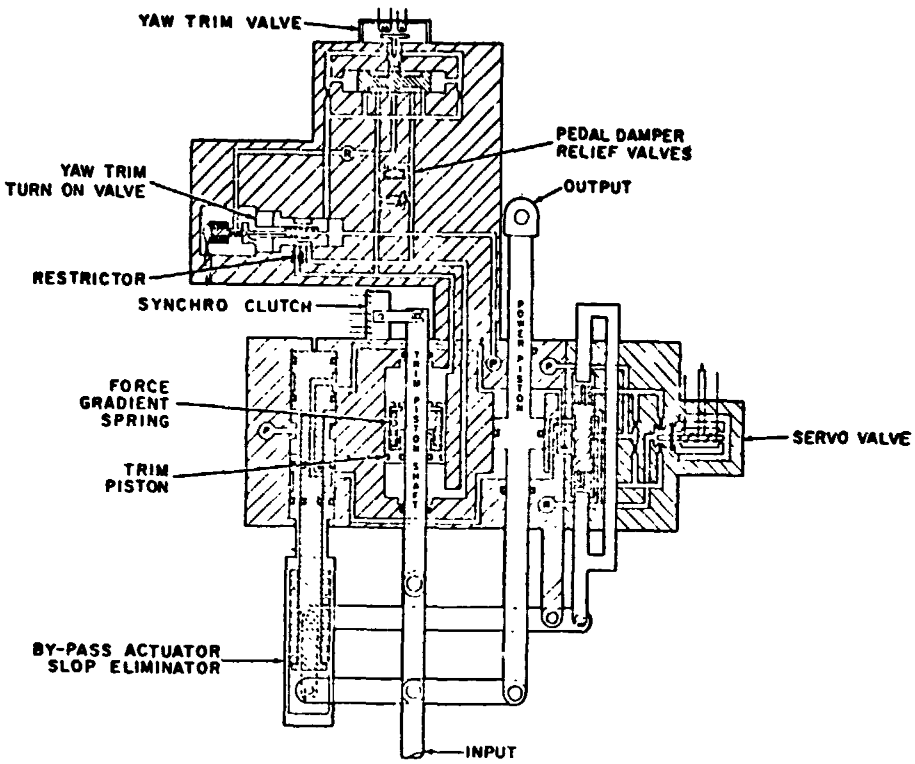


Figure 6.9 Yaw AFCS Servo Schematic

schematic, Figure 6.10, normally goes to the mixer and then to the tail rotor servo input. The mixer will be by-passed because the tail rotor collective pitch is independent of the main rotor collective in the HLA application. This can be accomplished by either physically by-passing the mixing unit or changing the mixing control horn to an idler which is not affected by the mixer.

The pedals will be rigged for the safety pilot such that in the forward helicopters depressing the right yaw pedal will result in thrust towards the starboard side. In the aft helicopters depressing the right yaw pedal will result in forward thrust. In the command helicopter the pilot's yaw pedals control differential pitch of the aft helicopter tail rotors and helicopter pitch attitude for HLA yaw control.

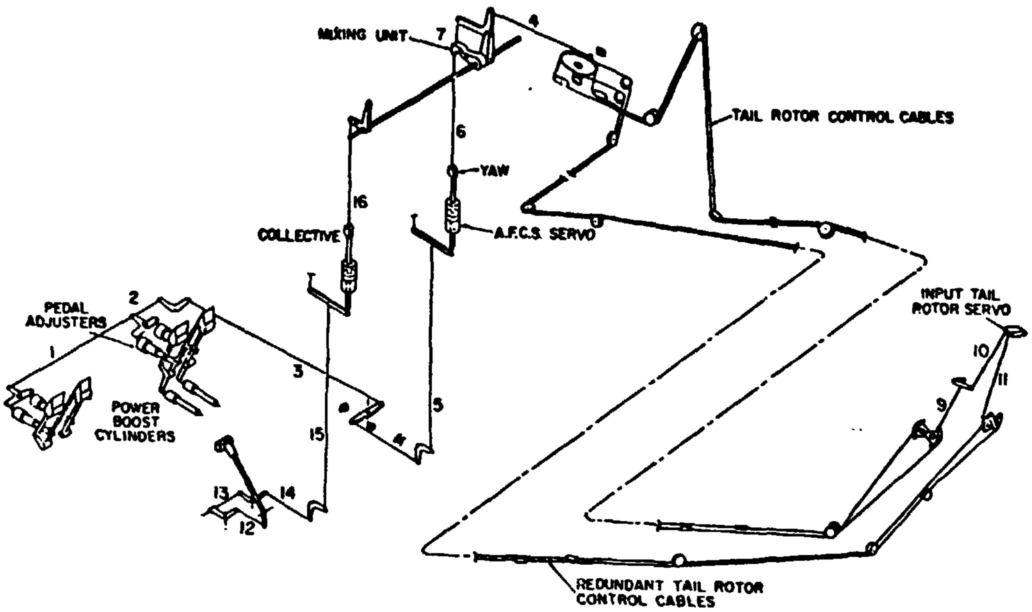


Figure 6.10 Tail Rotor Flight Controls Schematic

The yaw pedals will retain their adjustment feature for pilot comfort and the pedal dampers to limit the mechanical input rates to avoid overstressing the airframe.

6.9 Heavy Lift Airship Flight Control System

The fly-by-wire commands for all helicopters including the command helicopter are computed in the command helicopter via the control laws. Block diagrams of the control laws are shown on Figures 6.11 and 6.12. Figure 6.13 is a block diagram showing the main elements of the HLA flight control system and their interface with the helicopter's AFCS.

The command pilot has dual electrical cyclic and collective sticks and yaw pedals to generate the commands to the analog fly-by-wire flight control system where the control laws are implemented in dual paths for redundancy. This redundancy is continued to

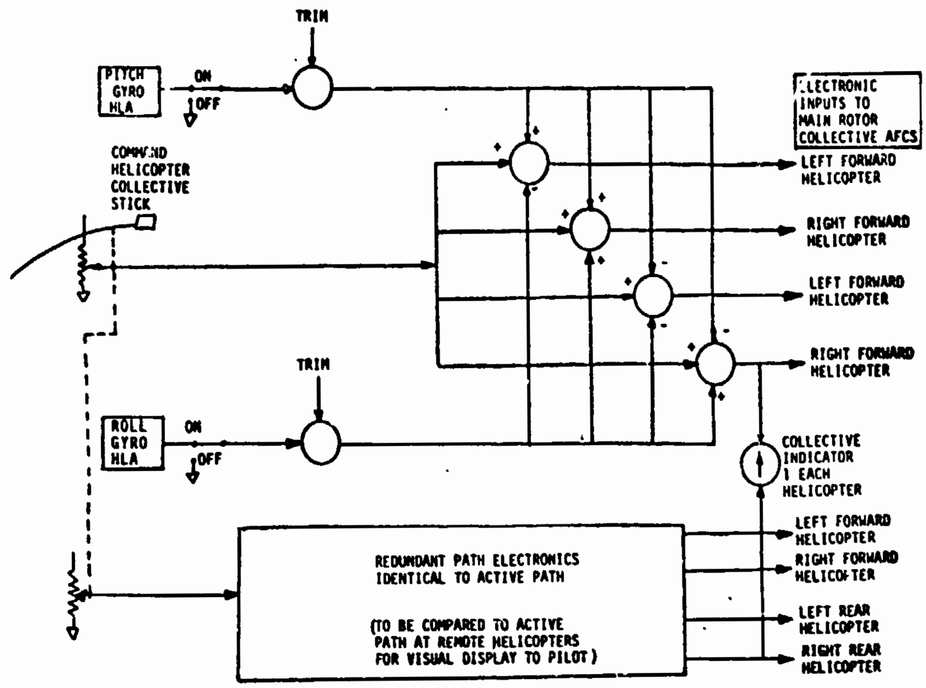


Figure 6.11 Main Rotor Collective Control Law/Autopilot

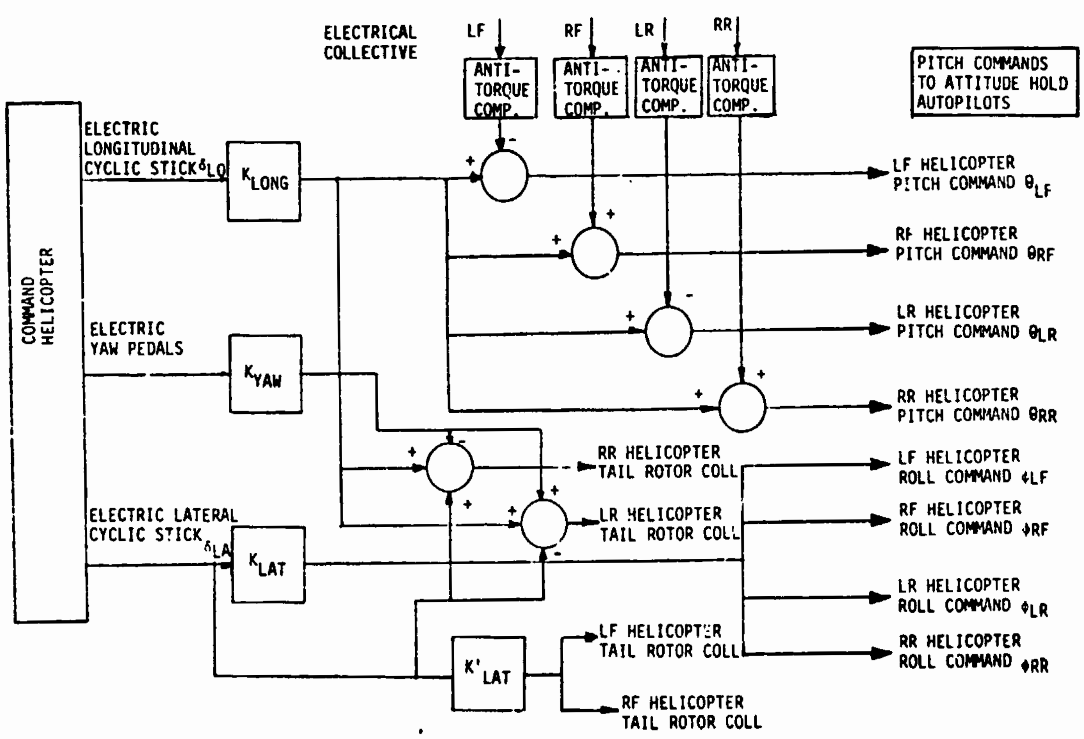
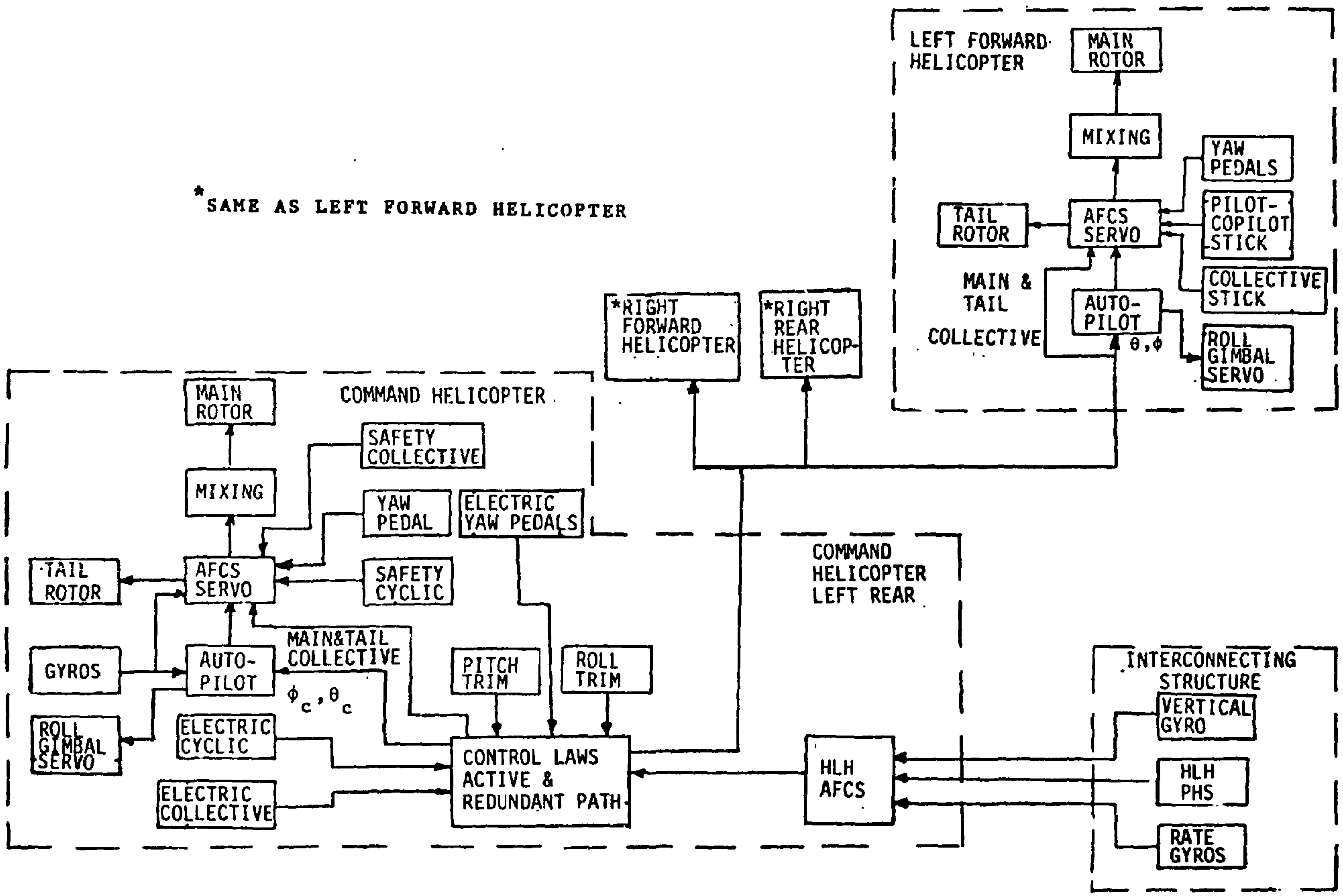


Figure 6.12 Main Rotor Cyclic and Tail Rotor Control Laws



*SAME AS LEFT FORWARD HELICOPTER

Figure 6.13 Heavy Lift Airship Flight Control System Block Diagram each helicopter and displayed to the safety pilot on a dual set of meters such that it can be determined if the channels are indicating the same command. This will give the safety pilot a visual indication that can be used to select either channel or average both channels. This is consistent with the CH-54B AFCS concept which has a redundant electrical channel such that either or both can be used. The HLA therefore has three modes of operation for safety-of-flight

(1) Active path fly-by-wire

(2) Redundant path fly-by-wire

(3) Safety pilot primary flight control system

The HLA fly-by-wire primary flight control system is a dual redundant system from the command pilot's commands to the input commands to each helicopter autopilot. The helicopter autopilot, also a dual redundant electronic system, flies the helicopter on the gimbal through the electro-mechanical AFCS servo. This servo drives the main and tail rotor pitch actuators. It accepts electrical inputs from the autopilot with a mechanical safety pilot's stick override capability. There is no electrical redundancy in the AFCS compatible with the HLA fly-by-wire concept. To provide redundancy, the fly-by-wire gimbal command signal is compared with the actual gimbal position signal on a meter which is visible to the safety pilot. When the helicopter is following the command signal, the meter needle is centered. If a malfunction occurs in the AFCS servo, the needle will indicate the failure, by a variance from the center position. The safety pilot can then override the AFCS and "fly-the-needle" (keeping the needle centered) to provide dual redundancy control.

If both fly-by-wire systems should fail the safety pilots will make an emergency landing under the command pilot's supervision as soon as possible using voice commands from the command pilot. It should be noted that an operational configuration utilizing a centered control car would use a triple redundant FBW control similar to that developed and demonstrated for the HLH.

Included in the command pilot's primary flight control system is an autopilot function which maintains the HLA in a level or constant pitch and roll attitude. Roll control is used only for the precision hover mode close to the ground. Normally the metacentric restoring moment is sufficient to maintain good roll control for most flight modes. These functions are accomplished by differential collective pitch of the main rotors and the main reason for making it automatic was to not increase the command pilot's work load. The command pilot has trim capability for both roll and pitch attitude. This system uses a vertical gyro sensor located on the interconnecting structure. The gyro output goes to the FBW electronics where the main rotor differential collective pitch and roll signals are generated.

The primary FBW flight control system provides the command pilot with the capability to perform all the flight requirements except precision hover which must be an automatic function.

6.10 Precision Hover

This mode is required to accurately position loads in gusty weather. The key element to precision hovering is a sensor to measure position and rate of the HLA relative to a desired position on the ground. Precision hovering was a requirement in the HLH development program. This included the development of a sensor and an AFCS with flight tests in a CH-47 Chinook to demonstrate feasibility.

The precision hover sensor for the HLH developed by RCA has some unique characteristics. At hover position initiation, the scene below is stored electronically and used as a reference. All future scene information is continuously compared to the original reference by a correlation-tracking device developed by Goodyear. This develops the horizontal X and Y position errors for inputs to the AFCS. The Z information is obtained by a precise pulse sine wave - modulated laser beam which can penetrate dust stirred up by the rotors. The entire sensor is gimbal stabilized to remove errors due to attitude of the helicopter or HLA in this application. This is an extremely accurate device weighing

500 pounds which measures position in all three axes to better than 1 inch from altitudes of 20 to 125 feet.

Operationally the pilot will not always obtain the exact location he desires when he initiates hover. He can then adjust his position without releasing and requiring a new lock-on, by operating his control stick in the following way. Intiially with the control stick centered, the HLA is motionless. When the pilot momentarily moves the control stick in the direction of the desired motion for less than 0.5 second the HLA moves 1 inch; which is defined as a beep. When the hover stick is in position for more than 0.5 second the system goes into a "creep" mode. Creep velocity is proportional to stick displacement.

Goodyear is presently developing advanced trackers using new solid state digital techniques and electronic stabilization that will permit the PHS weight to be reduced by an order of magnitude for the HLA operational application.

7.0 PERFORMANCE ANALYSIS

7.1 General

The performance of the HLA in both forward and vertical flight has been estimated. The performance has been defined for the following three conditions:

(1) Sea level*; 288.2°K (59°F)

(2) 1524 m (5000 ft) altitude; 278.3°K (41.2°F)

(3) 3438 m (8000 ft) altitude; 272.3°K (30.5°F)

The rotor performance is based upon the analysis of flight test data reported in Reference 7. These data were analyzed by Piasecki Aircraft Corporation during the Phase II Study for Goodyear. The results of this data analysis is included in Figures F-1 thru F-6 of Appendix F of Book II of this volume of the report. Provided in these figures for the above conditions

*Sea level herein refers to an altitude of less than 3048 m (1000 ft) is the power required versus gross weight including a breakout of the:

(1) Power required for the tail rotor with mechanical losses

(2) Power required for main rotor losses and accessories

(3) Main rotor profile power

(4) Main rotor induced power

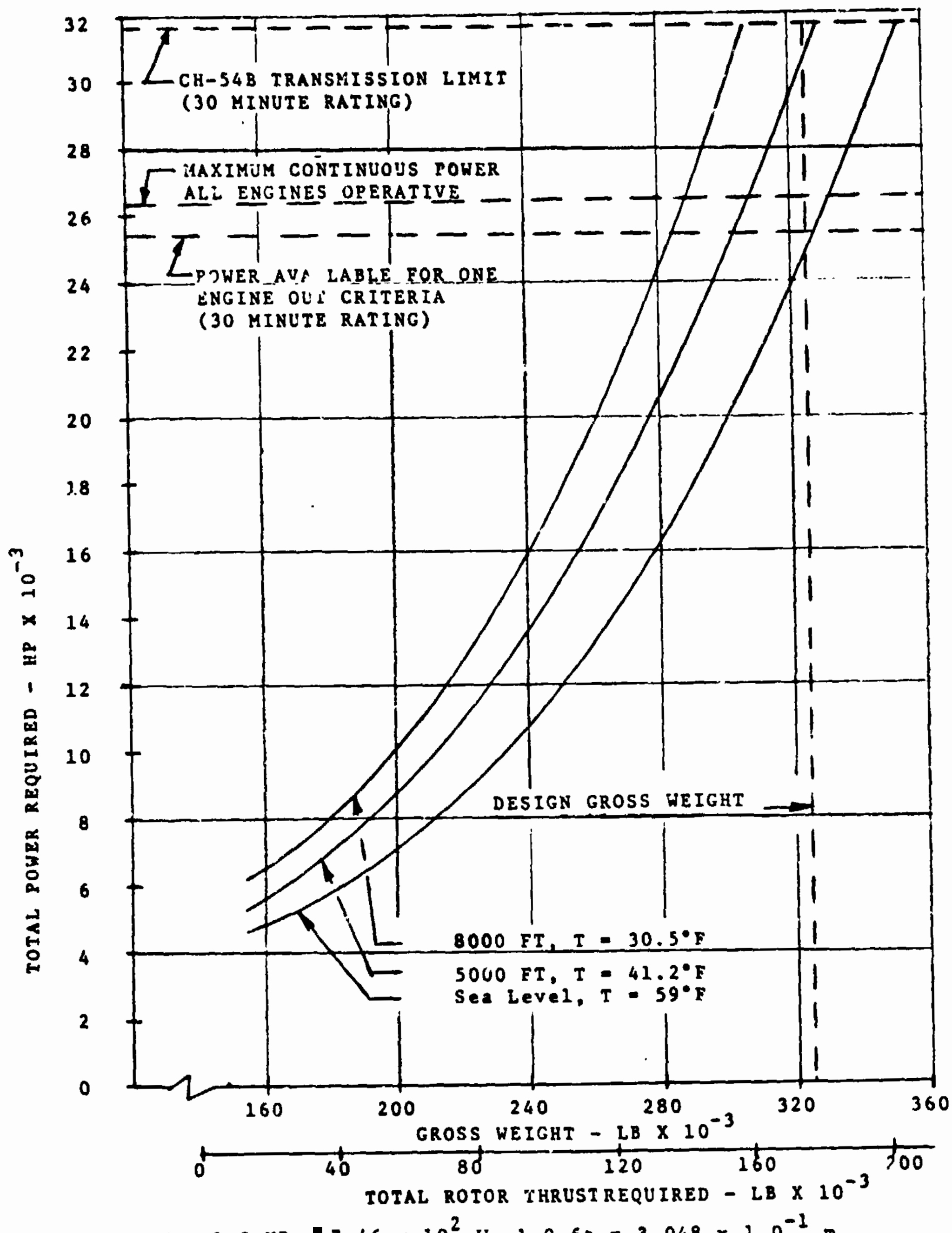
(5) Parasite power for non-zero airspeeds

The power required for helicopter thrust levels below 9070 kg (20,000 lbs) was approximated from data provided by Sikorsky Aircraft. These data are included in Figure F-7 of Appendix F of Book II of this volume of the report.

7.2 Vertical Flight

From Figures F-1 thru F-3 of Appendix F of Book II of this volume of the report and the available buoyant lift for the three conditions of Section 7.1, the power required versus gross weight curves of Figure 7.1 have been developed for Hover-Out-Of-Ground Effect (HOGE). Also accounted for in the curves of Figure 7.1 is the power margin required to permit a 30.48 m/min (100 ft/min) vertical climb capability.

As stated in Section 4.0, a design requirement for the vehicle is the ability to hover out of ground effect at sea level on a standard day at the design gross weight with one engine out with sufficient power for a 30.48 m/min (100 ft/min) vertical climb. As indicated in Figure 7.1 the one engine out power available is 1.89×10^7W (25,000 hp) which is in excess of the 1.83×10^7W (24,600 hp) required. The power required includes consideration for 11.11°K (20°F) of superheat. It should be noted that, historically, airship specifications have not reflected any allowance for the changes in buoyant lift associated with superheat effects. As a practical matter, however, superheat is an operational consideration and as a result has been included herein.



NOTE: 1.0 HP = 7.46×10^2 W, 1.0 ft = 3.048×10^{-1} m,
$t_K = (5/9)(t_F + 459.67)$, 1.0 lb = 4.54×10^{-1} kg

Figure 7.1 Power Required Versus Thrust Required and Gross Weight For Hover Out Of Ground Effect (With Sufficient Margin For a 100 ft/min Climb)

Generally, an allowance for 11.11°K (20°F) of superheat is adequate and accordingly this value has been considered. The one-engine out capability, of course, means that at the design gross weight, an engine failure will not require the payload to be released. Thus, both a safety hazard and a possibly significant economic risk are avoided. A one-engine out condition will require essentially maximum rated power in all remaining engines.

As indicated in Figure 7.1 the gross weight capability of the Phase II configuration at sea level and 288.2°K (59°F) with all engines operative at maximum rated power (transmission limit) is approximately 160,993 kg (355,000 lbs). As also noted in Figure 7.1, the design gross weight can be maintained at 1524 m (5,000 ft) with all engines operative at essentially maximum rated power (transmission limit). For operation at 2438 m (8,000 ft), the maximum gross weight that can be achieved is slightly over 136,050 kg (300,000 lbs). For the 2438 m (8,000 ft) altitude condition, the 11.11°K (20°F) of superheat capability is somewhat reduced.

7.3 Forward Flight

The power required for the HLA as a function of forward speed was estimated based on:

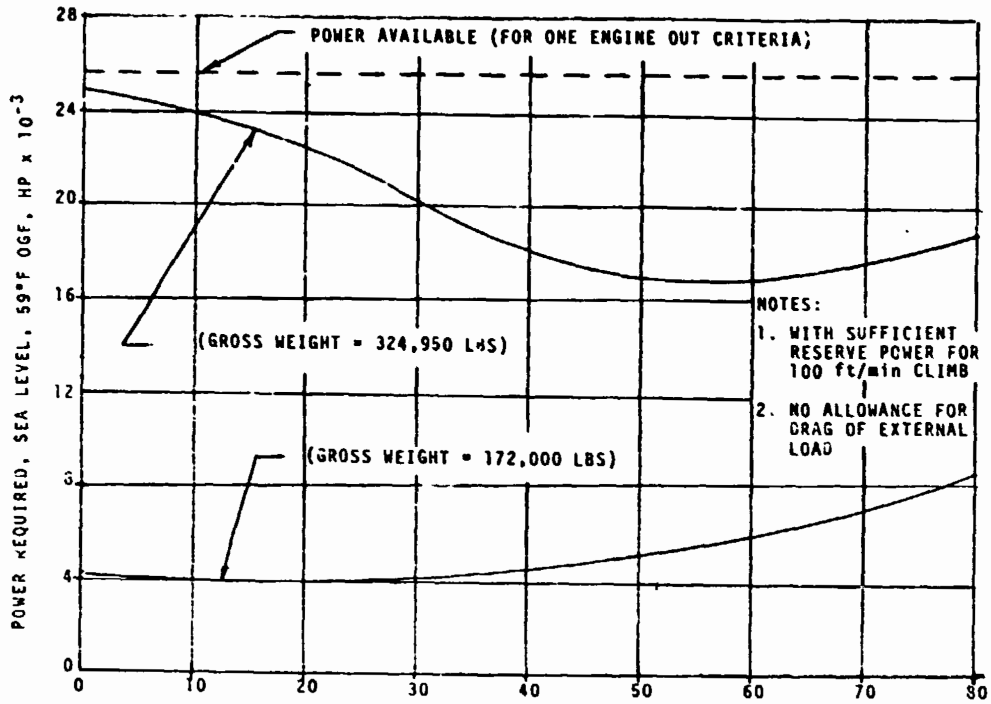
(1) The main rotor performance data of Figures F-4 thru F-7 of Appendix F of Book II of this volume of the report.

(2) The power absorbed by the aft tail propellers.

(3) An overall vehicle axial force coefficient of 0.082 (based on $\forall^{2/3}$) at $\alpha = \beta = 0$ derived as indicated in Section 5.11.3.2.2.

(4) Sufficient power margin to permit 30.48 m/min (100 ft/min) vertical climb.

The power required then consists of main rotor induced, main rotor profile, parasite (hull + struts + helicopters), tail propellers, mechanical losses and accessories, and that required to climb. Figure 7.2 provides the estimated power required versus



NOTE: $t_K = (5/9)(t_F + 459.67)$, 1.0 $H_P = 7.46 \times 10^2$ W, 1.0 lb = 4.54×10^{-1} kg, 1.0 ft/min = 3.048×10^{-1} m/min

Figure 7.2 Shaft Horsepower Required versus Airspeed

forward speed for both the design gross weight and the gross weight of the vehicle excluding any payload. Although no allowance has been included for the drag of the payload (since different payloads will have different drag characteristics), payload drag typically will be small compared to that of the vehicle.

7.4 Range

From Table 7.1, for the design gross weight at the design velocity at sea level and 288.2°K (59°F), approximately 10,214 kg (22,522 lbs) of fuel and oil is required to achieve the design range of 18,520 m (100 n m). This fuel requirement is essentially the limit of the standard tanks on the helicopters. This same fuel load results in approximately a 37,040 m (200 n m) range when the vehicle is flying without payload.

Figure 7.3 illustrates the ferry range (payload replaced with fuel) of the Phase II HLA vehicle for a cruise speed of 30.86 m/s

TABLE 7.1 PERFORMANCE SUMMARY

	SEA LEVEL 59°F	5000 FEET 41.2°F	8000 FEET 30.5°F
GROSS WEIGHT (LBS)[1,2]	324,950	306,000	287,500
EMPTY WEIGHT (LBS)	148,070	148,070	148,070
USEFUL LOAD (LBS)	176,880	157,930	139,430
FUEL AND	22,522	21,314	19,255
CREW	1,000	1,000	1,000
PAYLOAD[3]	153,358	135,616	120,175
STATIC HEAVINESS (LBS)	3,920	20,660	33,215
PERCENT INFLATION (AT SEA LEVEL)	0.930	0.822	0.77
PRESSURE HEIGHT (FT)	1,000	5,000	8,000
SUPERHEAT ALLOWANCE (°F)	20	20	8
DESIGN SPEED (KTS-TAS)	60	60	60
RANGE (NAUTICAL MILES			
DESIGN (WITH MAX. PAYLOAD)[4]	100	100	100
FERRY[5]	1,150	1,090	950
WITH NO PAYLOAD[4]	196	195	193

[1] Sufficient power available for 100 ft/min vertical climb with one engine out

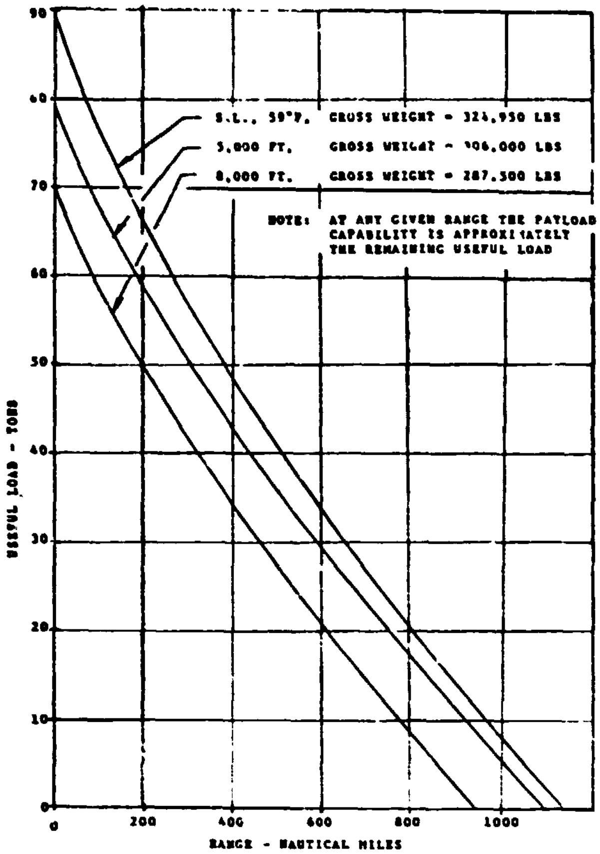
[2] Lift of helium is defined as 0.062 lbs/cu ft at sea level on standard day at 94% purity

[3] At design range of 100 nautical miles. Based on current empty weight estimate the payload capacity is somewhat greater than 75 tons

[4] Fuel load as defined above

[5] Useful load is all fuel except for crew

NOTE: 1.0 lb = 4.54×10^{-1} kg, 1.0 ft = 3.048×10^{-1} m, $t_K = (5/9)(t_F + 459.67)$, 1.0 kt = 5.14×10^{-1} m/s, 1.0 nm = 1.85×10^{2} m, 1.0 ft/min = 3.048×10^{-1} m/min



NOTE: 1.0 Ton = 9.07×10^2 kg, 1.0 ft = 3.048 $\times 10^{-1}$ m,
1.0 nm = 1.85×10^2 m, $t_K = (5/9)(t_F + 459.67)$

Figure 7.3 Useful Load Versus Ferry Range

(60 kts). These data are provided in such a manner that an approximate indication of payload capability can also be secured as a function of range. From Figure 7.2 it can be see that for the design gross weight the cruise velocity resulting in maximum range is something greater than 41.15 m/s (80 kts). The optimum speed for maximum range decreases as the gross weight decreases as can be concluded from the lower gross weight curve of Figure 7.2. For instance, 10,214 kg (22,522 lbs) of fuel at this lower gross weight curve results in a 36,299 m (196 n m) range which suggests the ferry range at the design gross weight, if optimum cruise conditions could be achieved at no increase in structural weight, would be considerably greater than indicated in Figure 7.3. The optimum speed for maximum ferry range will vary during a mission as the gross weight is changing. It will be a maximum at the start of the mission, approximately 46.30 m/s (90 kts) for the Phase II HLA, and decrease as the gross weight decreases. As

a zero fuel condition is approached (minimum flying gross weight) the optimum cruise speed will be a minimum for a given vehicle.

The increase in structural weight to achieve a 46.30 m/s (90 kt) cruise condition for the Phase II HLA has not been defined at this point. Considerable increase in forward speed is possible with very little increase in structural weight, however. It is clear that subsequent study should better optimize the HLA configuration in this respect. Directional control becomes a consideration at higher design speeds with the possible necessity of decreasing the cruise speed back to 30.86 m/s (60 kts) as severe turbulence is encountered. Structurally, however, the configuration would be designed to withstand the turbulence.

7.5 Performance Summary

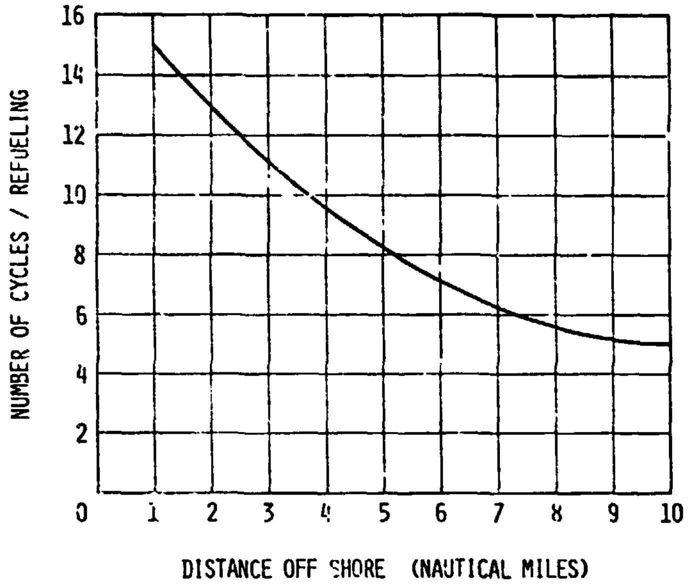
Table 7.1 provides a performance summary for the Phase II HLA. In all cases the payload capacity can be increased at the expense of range.

7.6 Logistics Over The Shore Mission Performance (LOTS)

One military mission examined from a performance standpoint was the unloading of cargo from off-shore ships. The mission parameters considered are those which were delineated in Secti 3.3.1. Figure 7.4 presents the results of this performance analysis in the form of off-loading cycles per refueling as a function of off-shore distance with the vehicle initially at the design gross weight.

Ferry Capability

As noted previously, the LOTS mission requires a ferry capability. As indicated in Figure 7.3 the unrefueled ferry range of the Phase II configuration at the design gross weight at sea level on a standard day is approximately 203,720 m (1,100 n m). Refuelings of airships at sea from tankers has been adequately demonstrated and suggests such an approach operationally for the HLA class of vehicles. Thus, the refueled range can be extended to essentially global distances.



NOTE: 1.0 nm = 1.85×10^2 m, $t_k = (5/9)(t_F + 459.67)$

Figure 7.4 Number of Complete Cycles/Refueling For LOTS Mission At Sea Level and 59°F

8.0 OPERATIONAL ANALYSIS

8.1 General

As indicated in Figure 2.1, Task II of the Phase II Study consisted of an operational analysis of the HLA. In originally assessing recent concepts combining buoyant and rotor lift the advantages offered in one area by a certain aspect of the concept were traded against the disadvantages that this same design aspect created in another area. The HLA concept studied herein, as well as other recent concepts in this area, permit consideration of buoyant lift to more than offset the empty weight of the vehicle. There is, from a purely economic standpoint, a benefit to be derived from lifting a portion of the useful load by buoyant lift. This requires, however, that rotor thrust be used to offset the aerostatically light condition when no and/or a light load condition exists except when moored. This situation creates a most demanding operational requirement in terms of ground handling and mooring or launch and recovery as it is sometimes referred to in discussing aerostatically light concepts. Given the problems encountered in the past with large neutrally buoyant airships over the range of environmental conditions that realistically must be considered, it is believed that "statically light" concepts may result in severe operational problems and restrictions. The above decision offers the possibility of significantly reducing past difficulties in handling airships once on the ground. It is anticipated that the rotor systems on an operational HLA vehicle would be designed to permit a substantial down thrust capability to enhance the taxi characteristics on the ground. This capability can be achieved with the FRV although some basic modification to the helicopters would be required.

8.2 Normal Flight Procedures

8.2.1 General

Chapter 3 of Reference 13 delineates the pre-flight, flight and post-flight procedures for the CH-54B helicopters. As indicated in Reference 13, when the CH-54B is flown on a regularly scheduled basis by the same flight crew only a portion of the normal required checks are necessary. It is anticipated that a similar approach can be followed with the HLA. The normal flight procedures for the CH-54B will be combined with LTA related procedures. In addition to the required checkout of a limited number of LTA related subsystems the following would be expected to require pre-flight procedural attention:

(1) Static lift conditions

(2) Static trim conditions

(3) Superheat variations

The above require elaboration because although similar considerations were involved in past LTA operations the presence of large amounts of rotor lift significantly diminish their importance in the HLA concept. For normal day-to-day operations when flight near the design gross weight is not anticipated the exact amount of static lift available will not be a critical parameter in that rotor lift can be used to offset the absence of buoyant lift. For flight at or near the design gross weight it will be necessary to know with reasonable accuracy the amount of static lift avaiable in order to assure that the one engine out condition can be satisfied. Excluding this condition the most significant reason for knowing the static lift condition on a continuing basis is to avoid prolonged helium loss that might occur in visually unobservable areas due to envelope damage. Weigh-offs, in the sense of past LTA practices, cannot be performed with the HLA due to its static heaviness. It is anticipated that the amount of static lift will be determined by measuring the hydraulic pressure in the individual landing gear oleos. These readings would be relayed to the command helicopter and converted to the total load on the landing gears. Knowledge of the empty weight of the vehicle plus onboard fuel, oil, crew, and provisions would then permit the buoyant lift to be determined for landing gear measurements made in a no-wind condition. These measurements will be made on a continuing basis when the no-wind conditions exist and a log kept. Prior to flight, the crew would refer to the curreng log and thereby not delay the flight awaiting a no-wind condition.

The ballonets will be used to effect static trim for cruise flight when weight and balance characteristics permit. Rotor thrust will be used as required to augment the ballonets in maintaining pitch trim. As noted in Section 5.10, the pitch autopilot will control the ballonet conditions and the individual rotor thrust levels to maintain trim in pitch.

As indicated in Table 7.1, the performance of the vehicle for the design gross weight is based upon a 11.11°K (20°F) allowance for superheat. With a minimum static heaviness of 1778 kg (3,920 lbs) and the ability to generate negative thrust, superheat will not present operational difficulties.

8.2.2 Preflight and Taxi Operations

Preflight vehicle checks will be made with the vehicle moored. Following completion of the pre-flight procedures and rotor engagement, the vehicle will taxi from the mooring point to the takeoff area with takeoff generally accomplished in a nose into the wind attitude.

The minimum static heaviness plus the additional heaviness which will occur due to the crew and at least several thousand pounds of fuel will permit the vehicle to taxi using the thrust of the aft tail rotors. Negative pitch on the tail rotors will permit the HLA to move rearward as required on the ground. The rudder pedals will provide directional control when taxiing. Down thrust in the operational vehicle can be used to augment the heaviness condition but should not be required under most taxi conditions. Another feature of the HLA favoring improved characteristics when on the ground is the wide based landing gear arrangement. Thus, while moderate ground winds can produce rather large side forces and substantial overturning moments, the wide based landing gear arrangement can be expected to handle any conditions under which taxi operations would be conducted. Of course, the tolerance of past airships to crosswind conditions when on the ground and not moored was minimal due to a narrow based or single wheel landing gears.

There is no intention above to suggest that the HLA will require extensive taxi maneuvers as a normal procedure but when it is necessary or even convenient to perform such ground maneuvers the HLA provides a dimension in this regard heretofore impossible with LTA vehicles.

8.2.3 Takeoff; Cargo Pickup and Placement; Landing Operations

Normal takeoff will be vertical with the nose into the wind, the pilot performing essentially the same functions as in the case of a helicopter. Transition into forward flight will also require pilot functions similar to those for a helicopter. Upon arrival at the load pickup point a nose into the wind attitude will be assumed over the pickup area. As is typical in helicopter operations, the load will generally be attached to the cargo handling system by ground personnel in accordance with predetermined attachment arrangement compatible with vehicle c.g. limitations. The load will be preslung when possible which will significantly facilitate the load acquisition phase. It is anticipated that the precision hover system will be used in acquiring and placing essentially all cargo. This in effect means that the command pilot will only have to regulate altitude through collective pitch control when acquiring and placing most cargo. If a condition is encountered during load acquisition in which station cannot be maintained, the pilot will assume control and reacquire the desired hover position and again "lock on" with the precision hover system. Once the load is acquired, the vehicle will climb vertically and transition to forward flight.

The HLA will be similar to a helicopter on the ground and in the air once the rotors are engaged. Prior to rotor control forces being available the vehicle will be similar to an airship. Accordingly, as suggested earlier, the vehicle will remain moored up through the point of rotor engagement.

8.2.4 Refueling

In those missions requiring ranges beyond those of Table 7.1, in-flight refueling from a surface vessel may be necessary. The HLA with its improved hovering qualities, with respect to past airships, will greatly facilitate the refueling operation. Refueling could be effected from tank truck, tanker ship, or permanent ground facility.

8.3 Emergency Flight Procedures

Current FAA standards for Category A rotorcraft require that following a one engine failure during takeoff at the design gross weight and without the release of payload or fuel that a safe landing at the takeoff area be made or the flight be continued. Elsewhere in the flight profile it is required that the flight continue to a safe landing area. As in Reference 1, this FAA requirement has been adopted for the HLA vehicle which in effect necessitates designing, as noted earlier, around two engines being inoperative in order to maintain trim.

Emergency procedures for the CH-54B (Chapter 4 of Reference 13) indicate that under certain gross weight and altitude conditions level flight can be maintained after partial loss of power. In many cases if a single engine is lost and cannot be restarted, an immediate landing is necessary which is commonly cushioned by application of full collective pitch control just prior to impact. It is often necessary to release externally slung cargo in order to make a safe single engine landing. In contrast, the one engine out condition will not require release of the payload for most conditions in the case of the HLA.

Failure of more than one engine is considered very remote and the FAA has no requirement on the subject of multi-engine failure for Category A rotorcraft. Should this unlikely situation occur in one helicopter, the HLA is capable of continuing flight and effecting a safe landing. For this situation there would still be circumstances and gross weight conditions which if not exceeded would permit the cargo to be placed intact on the ground adjacent to the area where the landing was planned.

The only other emergency situation comparable to engine power loss in terms of safety of flight would be the loss of a main rotor. The possibility of losing a main rotor or rotor blade in a non-combat environment is remote, based on available statistics. Loss of a rotor may be no different in terms of effect on the capability of the HLA to maintain flight and/or land safely than loss of complete power in one helicopter. In the case of a rotor or portion of a blade contacting the envelope in flight, it is unforeseeable that damage to the envelope would be so severe as to prevent a safe landing of the vehicle.

8.4 Weather Flight Procedures

8.4.1 General

The HLA offers considerable improvement in comparison to prior LTA vehicles in adverse weather conditions.

8.4.2 Wind

Wind has historically been the most important weather element for LTA vehicles and has required special flight procedures. The limited forward speed of past airships necessitated that high head winds be avoided by flying the pressure patterns. Many prior airship operations involved extensive range requirements where this high head wind situation was a severe problem. The heavy lift mission will generally be short and in some cases will be repetitive such that the wind will be beneficial during a portion of the mission as well as detrimental during a portion. Thus, while the design speed considered for the HLA during the Phase II study is not particularly high this will not present the same limitations that it did in prior airship operations. As noted in Section 7.0, the design velocity of the Phase II HLA can be increased perhaps as much as 25 percent without appreciably affecting the structural weight. Additional analysis of the vehicle mission requirements is required to select the best design velocity. With the exception of ferry missions, head winds will not require special flight procedures.

In comparison to prior airships, sensitivity to other in-flight wind conditions is greatly reduced for the HLA. Whereas prior LTA vehicles have not had a hover capability under variable wind and superheat conditions, the HLA has very substantial capabilities in this respect. It is clear, from the wind tunnel and 6 DOF simulation results, that the available rotor forces in combination with the precision hover sensor permit the HLA to hover, takeoff, land, and to pickup and place cargo under comparatively adverse turbulent conditions. As such the procedural requirements for both the flight and ground crew will be minimal in comparison to prior LTA operations.

8.4.3 Cold Weather Operation

As reported in Goodyear's Phase I report (Reference 14), in-flight ice and snow accumulation is not a problem with airships given proper preventive measures in a few critical areas. In cold weather, when operating from a snow covered surface, blowing snow from rotor downwash could present visibility problems. This condition would be improved in subsequent configurations in which the pilot would be in a central control car thus displaced from the rotor downwash. Landing on loose snow may present some difficulty; however, there are techniques developed for helicopters that will be entirely adequate for the HLA. It will be necessary to ensure that all ice and snow are removed from the rotor blades in preparation for flight as in the case of any helicopter. Landing gear, actuators, etc., must also be checked to assure that ice and snow accumulations would not prevent their normal functioning.

The CH-54B is not qualified for flight in icing conditions. This is not a serious drawback for a flight research vehicle because operation in these conditions is not required.

8.4.4 Salt Water Operation and Other Environmental Factors

Hovering at certain altitude and ambient wind combinations in the presence of salt spray can result in eventual power losses in the helicopter if the practice is continued. The procedures applicable to the helicopters will govern the overall vehicle operations in this area. The interconnecting structure will be protected from all environmental factors. The envelope is generally insensitive to all environmental effects with the exception of ultraviolet radiation. In this respect it has been necessary to apply what is termed a wash coat to the top side of the envelope on a yearly basis. This subject is discussed in greater detail subsequently in Section 8.8.

8.4.5 Turbulence and Thunderstorm Operation

It has been typical in past airship operations to avoid thunderstorms; however, experienced pilots have shown that properly designed airships can safely fly in this environment.

In general, the HLA will be operated in such a manner as to avoid heavy turbulence. Current avionics and communications equipment on board the CH-54B coupled with current long range weather forecasting capabilities will permit significant improvement in the successful procedures developed in the past.

Lightning has never caused concern with a helium-inflated airship. Although all aircraft attempt to avoid lightning areas because of the turbulence that usually exists, there has been evidence of strikes on airship cars, fins, and topside radomes but none that caused detectable damage to an envelope of a non-rigid. There have been reports of small holes in the outer coverings of rigid airships where lightning hit the metal structure beneath, but the structure beneath was not damaged. No special operational procedures are envisioned in this respect with the exception that thunderstorm areas will generally be avoided.

8.5 Ground Handling and Mooring

8.5.1 Operational Base/In-Field

At the operational base there are many reasons a paved mooring circle and take-off area would be desirable. The current center point mooring concept greatly reduces the amount of real estate required from which to operate the HLA in comparison to prior airships. There will be no need for hangar facilities from an operational standpoint. The mooring system at the operational base will permit the HLA to be moored out under all but the most infrequent high wind conditions. There will be cases such as when hurricane type conditions are forecast when it will be necessary to fly the HLA from the area similar to the practice followed with heavier-than-aircraft when hangar facilities are not available.

An expeditionary mooring system can be transported with the HLA and easily erected adjacent to the area where the work is to be performed. As illustrated in Section 5.0, the helicopter alighting gear is suited for use in the HLA application. The footprint pressure of the HLA will be considerably less than that of the helicopter alone because due to the buoyancy of the HLA the weight supported by the four helicopter gears is considerably less than that of a single helicopter. As a result, a reasonably wide range of soil conditions can be considered in selecting a temporary mooring area. The local soil conditions which are selected will have an influence on the wind conditions in which the vehicle can be moored.

A ground crew will not be required to assist in either takeoff, landing, or mooring operations. The crew that normally maintains the vehicle will assist in pre-flight and post-flight checkout of the vehicle similar to the approach used in current helicopter operations. The vehicle will not be released from the mooring mast either at the operational base or at the temporary in-field base of operations without the ro - systems engaged. As a result ground support vehicles, mobile masts, etc., will not be required at these areas.

The helicopters will require consideration from a mooring standpoint much similar to their existing individual requirements. It will be necessary to restrain the rotor blades to the interconnecting structure as opposed to the ground due to the movement of the HLA with wind shifts when moored.

8.5.2 <u>At Point of Manufacture</u>

At the point of erection and final assembly, a hangar facility and ground handling vehicles will be required. In general the HLA will be removed from the hangar at the point of manufacture with the rotors folded back (or removed) and transported by ground handling equipment to a permanent center point mooring cup. Once at the permanent mooring cup, the vehicle will be brought to operational readiness for final acceptance.

The procedures for docking and undocking and ground handling the HLA at the point of manufacture will be similar to past airships but in general less difficult because of the static heaviness and wide based landing gear arrangement. Goodyear has developed, as a portion of its efforts relative to the HLA, ground handling equipment and vehicle concepts for use with the HLA during the recommended FRV program. In general this equipment is considerably simpler than prior ground handling equipment. The HLA will require a tractor for motive power and a second tractor to control the yaw attitude of the vehicle during docking and undocking as well as dollies for attaching the tractors to the HLA interconnecting structure.

8.6 Cargo Handling Procedures

It is anticipated that a wide range of payload sizes and weights will ultimately be transported by the HLA. As with tandem rotor helicopters, only to a greater extent with the HLA, longitudinal c.g. variations will not present the problem that it does with single rotor systems. In the case of the HLA, lateral c.g. variations will be much more tolerable due to the lateral spacing of the rotors as well as the metacentric restoring moment of the airship. The exact manner in which the cargo will be suspended will include consideration of many factors (e.g. weight, size, hard points, distance to be transported) including constraints that the payload itself will impose.

It is anticipated that a tandem cargo handling system similar to the HLH (see Figure 10.6) will eventually be desirable and accordingly it has been recommended that the existing HLH cargo handling system be evaluated on the HLA FRV. For transport of certain cargo over considerable distances it may be desirable to utilize payload stabilizer lines which would minimize payload dynamics. Much applicable data exists in general from helicopter experience that will serve as basic guidelines for the HLA in the area of acquiring, securing, transporting and releasing cargo. Further development of the flight dynamics simulation, as recommended in Section 10.5.4.1, to include the capability to model the dynamics of the payload, will also enhance the development of adequate cargo constraint techniques. Of course, the recommended research vehicle will serve as the major tool in developing adequate cargo handling techniques.

It is anticipated, based upon simulation studies, that the precision hovering qualities of the HLA will be adequate to perform the majority of heavy lift missions. In missions requiring greater precision, ground lines can be considered where practical.

A typical cargo pickup and placement procedure would be similar to that described below:

(1) Vehicle hovers over payload and nose into the wind where possible. The vehicle can land beside cargo for hook up if desired and conditions permit. In either case, vehicle is airborne before payload is lifted.

(2) Payload is generally preslung to minimize hookup time.

(3) Payload is generally attached to vehicle by ground crew.

(4) Payload stabilizer lines are attached if required by ground crew to minimize payload dynamics.

(5) Vehicle rises vertically and transitions to forward flight.

(6) At destination HLA transitions to hover mode with nose into wind where possible.

(7) Vehicle hovers while positioning payload at desired location. Vehicle can place payload in final position and land beside it prior to disconnect if desired and conditions permit.

(8) Ground positioning lines can be used as required and as practical to perform any final cargo positioning adjustments or to augment precision hovering capability of vehicle.

(9) Cargo is released and payload stabilizer lines disconnected either from vehicle or from the ground.

8.7 Personnel Requirements

8.7.1 General

In developing the personnel requirements for the HLA it was necessary to postulate an operational scenario. A commercial scenario was considered although a similar approach would conceivably be applicable to a military operation. The scenario discussed below formulates the basis for the total operating cost analysis (TOC) developed in Section 9.0 of this report.

It is envisioned that a commercial operation of the HLA would be similar to the commercial use of crane helicopters. The operation of the HLA will require a base of operation including a paved mooring area and center point mooring cup as illustrated in Figure 5.21. An operations building and maintenance facility will also be required. It is currently envisioned that the maintenance facility will be sufficiently large to accommodate one of the helicopters. Major maintenance of the envelope would be performed by the manufacturer for the operator because of the requirement for a dock facility.

The operator would not require ground handling equipment because the HLA will not be moved from the mooring cup unless the rotor thrust is available for vehicle control.

The TOC analysis of Section 9.0 considers the operational HLA configuration illustrated in Figure 1.2. The operational configuration differs from the FRV in that only a rotor/turbine module is retained on each outrigger and a central control car is provided for the flight crew. It is estimated that this approach would result in approximately a 20% reduction in the acquisition cost of the Phase II configuration which retains the entire helicopter on the outriggers. The flight crew requirements discussed below are also for the operational concept. The only detailed maintenance requirements data available during the study were for the S64F which is the commercial version of the CH-54B helicopter. As a result, the maintenance personnel requirements listed below and the maintenance cost requirements for the helicopter related components of the operational HLA configuration (e.g. rotor, turbine, transmission) listed in Section 9.0 are based on the S64F data. Since maintenance requirements for current technology helicopter components are considerably less than those for the CH-54B, the TOC's of Section 9.0 are conservative for a new technology dedicated rotor/turbine module of the type which would be used on an operational vehicle.

8.7.2 Rotor/Turbine Module Components and Subsystems

Typical maintenance personnel requirements for one S64F are:

	Utilization (Hrs/Annum) 1000	1500	1800
Chief Mechanic	1	1	1
Mechanics	4	4	4

Based on discussions with Sikorsky, it appears that approximately twice this number of personnel can properly maintain a vehicle with four rotor/turbine modules. The total personnel requirements were considered as:

	Utilization (Hrs/Annum) 1000	1500	2000
Chief Mechanic	1	1	1
Mechanics	8	8	10

8.7.3 Envelope and Related Components

The maintenance personnel requirements to perform routine in-field maintenance on the envelope and related components have been based on past experience with large non-rigid airships. Personnel requirements for the envelope and related subsystems are:

	Utilization (Hrs/Annum) 1000	1500	2000
Mechanics	1	1	2
Riggers	2	2	3

As noted earlier major maintenance on the envelope is assumed to be performed by the manufacturer for the operator.

8.7.4 Interconnecting Structure (IS)

The personnel required to maintain the IS were not directly estimated. The cost to maintain the IS was developed on the basis of the ATA approach for aircraft fuselage maintenance.

8.7.5 Flight Operations

It was assumed that a manager, assistant manager, and secretary would be required to support a typical commercial operation.

8.7.6 Flight Crew

Based upon the required flight crew activity during the normal short haul heavy lift mission the following flight crew requirements are believed reasonable:

	Utilization (Hrs/Annum)		
	1000	1500	2000
Pilots	2	3	3
Co-Pilots	1	2	3
Cargo Handling Operator	1	1	2

The flight crew requirements for ferry missions could be expected to vary from the above.

8.8 Maintenance Requirements

8.8.1 General

Maintenance of the HLA will be accomplished at the operational base at the mooring circle. When maintaining the individual rotor/turbine modules in adverse weather portable structures would be erected around them. The operational base will have a permanent maintenance facility in the operations building in which component maintenance can be performed.

Routine maintenance of the envelope and related LTA subsystems will be performed at the mooring circle for the adjacent maintenance facility. Major maintenance on the envelope will require return to the point of manufacture or other available dock facilities.

8.8.2 Rotor/Turbine Modules and Related Subsystems

As noted above the total operating cost analysis considered the maintenance requirements of the CH-54B in terms of retirement interval and overhaul interval as typical for the operational HLA. In conjunction with Sikorsky Aircraft, the required number of personnel to maintain the helicopter type components and subsystems were defined as discussed previously.

8.8.3 Envelope and Related Subsystems

The maintenance of the envelope and related subsystems is based on past experience with large non-rigids. The personnel requirements for normal or routine maintenance at the operational base were defined in the previous section. As noted previously, this routine maintenance would be performed at the mooring circle or adjacent maintenance facility at the operational base. The application of the wash coat to the top of the envelope to protect the fabric against ultraviolet degradation, however, will require annual docking of the HLA. The motion of the top of the envelope, even in minimal wind conditions, precludes considerations of conventional scaffolding at the mooring circle for this purpose. The use of ropes over the top of the envelope has not proved practical for this purpose. New materials (to man-rated airships) offer promise in removing the necessity of this requirement. These materials have other characteristics (e.g. foldability, flexibility, deformation capability) however, that require further investigation. The fabrics technology program recommended in Section 10 encompasses the investigation of methods and materials to remove the necessity of annual docking of the HLA. The cost of this particular maintenance item has been considered for purposes of the TOC analysis of Section 9.0 to be a service provided by the manufacturer to the operator on a contracted basis.

8.9 Institutional Constraints Analysis*

8.9.1 General

A thorough analysis of institutional constraints is normally based on a detailed knowledge of a specific market. A

*The results of this analysis are applicable to civil missions only.

detailed market study was, however, beyond the scope of the Phase II study and thus only a qualitative assessment of the institutional considerations is presented herein.

8.9.2 Route Structure

It is likely that the most severe constraint on the flight of the HLA, at least initially, will be that it may be necessary to route it over sparsely populated areas, river beds, etc. Certain ferry missions might require that the coast line be flown to avoid major metropolitan centers. Certain missions would require that the route structure allow for refuelings and emergency landings. These regulations, if initially instituted, may be lessened following the accumulation of successful flight experience with the HLA.

People along the route will be concerned with the ecology and environmental aspects of the passage of the HLA particularly if repetitive flights became necessary. Noise and propulsion system exhaust would be major concerns although inflight noise will be below current takeoff levels for CTOL aircraft.

8.9.3 Local Government Interactions

Operations within a local community or within a state become local issues. Even though the certification of the HLA is reserved for Federal agencies, the operation of such a system may have to meet the idiosyncrasies of the various local governments with respect to right-of-way, safety, licensing, and liability.

8.9.4 Federal Government Regulation Requirements

The prospect of a new transportation system places an especially unique burden on the Federal government agencies that are responsible for regulating the interstate transportaiton of cargo. There is a whole family of Federal regulatory agencies that become involved in establishing the regulations for its operation including:

- Economics (CAB)
- Equipment Certification (FAA)
- Operational Certification (FAA)
- Airmen Certification (FAA)
- Aeronautical Publications (FAA)
- Environmental Impact (FAA and EPA)
- Energy Conservation (FEA)

The FAA plays a major role in the introduction of any new airborne system. This agency will rely heavily on the technical judgment of its own technical staff, NASA, and contractor support in its assessment and regulation of this vehicle. This may be a lengthy process, primarily, because of the lack of precedents to draw upon and, secondarily, because of the educational aspects of the certification process.

The environmental impact of this system will concern both the FAA and EPA as well as the aforementioned local and state governments. The FEA in conjunction with the Energy Research and Development Agency (ERDA) will be concerned about the energy requirements of the new transport system.

8.9.5 Air Traffic Control System

The HLA will generally operate at low altitudes along predetermined routes under Visual Flight Rules (VFR). Because of the low altitude, the routes normally will be below the controlled Instrumented Flight Rules (IFR) airspace for commercial transports, therefore, flight planning and coordination with the FAA and ATC is currently not mandatory. It is anticipated, however, in cases involving operation in conjested airspace, such as airport terminal areas, that coordination with the above agencies will be required as it is in the current Goodyear airship operations.

8.10 Concluding Remarks

A majority of the elements in an operational analysis require actual experience in the field to arrive at the most acceptable procedures, regulations, etc.

9.0 ECONOMIC ANALYSIS[1]

9.1 General

An estimate of the total operating cost (TOC) per available payload ton-mile (statute) has been made as a part of the Phase II Modern Airship Study. The TOC analysis considers the operational configuration illustrated in Figure 1.2. The operational configuration differs from the FRV in that only a rotor/turbine module is retained on each outrigger and a central car is provided for the flight crew. It is estimated that this approach would result in approximately a 20% reduction in the acquisition cost of the Phase II configuration which retains the entire helicopter on the outriggers. Flight crew requirements and costs are also based on the operational configuration. As noted in Section 8.7.1, the only detailed maintenance requirements data available during the study were for the S64F which is the commercial version of the CH-54B helicopter. As a result the maintenance costs used in the TOC analysis for the helicopter type components of the operational HLA configuration (e.g. rotor, turbine, transmission) are based on the S64F data. This approach leads to a conservative TOC since the S64F/CH-54B maintenance requirements are considerably larger than current technology helicopters.

The TOC was considered to consist of the direct operating cost elements shown in Figure 9.1 and the indirect operating cost elements shown in Table 9.1.

9.2 Direct Operating Costs

9.2.1 Flyaway Cost

The DOC model of Figure 9.1 is the standard ATA approach. The cost of the initial operational vehicle was developed as indicated below:

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[1]Unless otherwise stated all dollars are constant 1976 dollars.

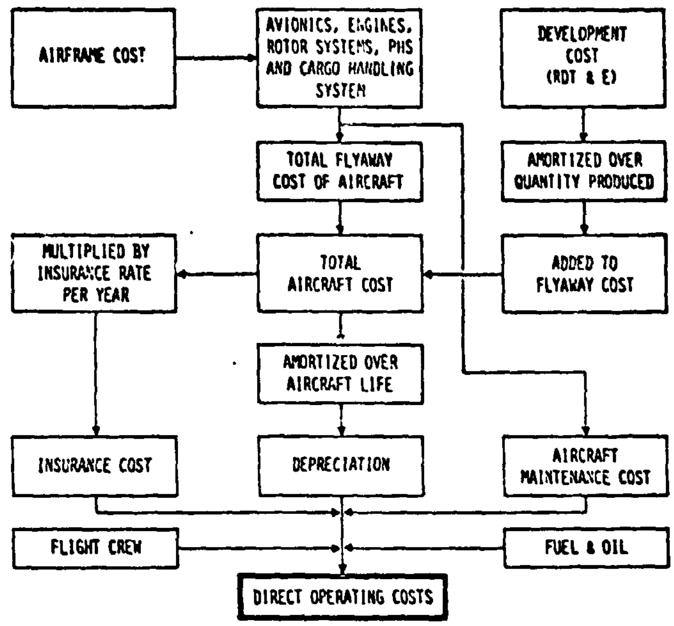

Figure 9.1 Direct Operating Cost Model

(1) Envelope Group

Manufacturing costs were based on historical data relative to the number of manufacturing man hours required per cubic foot of volume.[1] The fabric and other material costs as well as air system components were based on existing current airship cost data.

(2) Interconnecting Structure

Labor hours, material costs, and tooling costs for the interconnecting structure components and assembly were based on recent corporate estimates which are provided subsequently.

[1]In all manufacturing estimates a $12.00/hr rate was used as representative of the labor plus overhead rate for major airframe manufacturers.

TABLE 9.1 INDIRECT OPERATING COST ELEMENTS

I	OPERATIONS BUILDING, OPERATIONS EQUIPMENT, OPERATIONS SUPPORT VEHICLES, OFFICE EQUIPMENT, SHOP AND TEST EQUIPMENT, FUEL AND OIL STORAGE FACILITIES.
II	GROUND SUPPORT EQUIPMENT
III	GROUND HANDLING AND MOORING EQUIPMENT
IV	MOORING CIRCLE
V	MAINTENANCE AND MAINTENANCE BURDEN ON ITEMS I - IV
VI	GAS REPLENISHMENT
VII	REAL ESTATE TAXES
VIII	UTILITIES
IX	OPERATIONS MANAGEMENT AND ADMINSITRATIVE SUPPORT

(3) Rotor/Turbine Module Components and Related Subsystems

The module components and related subsystems include the rotor group, body group, alighting gear, rotor system controls, engine section or nacelle group, propulsion group, hydraulic and pneumatic group, electrical group and auxiliary power plant group. The cost of these elements was based on recent data for the S64F since the component requirements are very similar. The current flyaway cost ($1976) of the S64F in small quantities is $\$6.1 \times 10^6$. This compares to a flyaway cost for this model in 1969 of $\$2.7 \times 10^6$.

(4) Control Car

An estimate of the control car weight was developed based upon crew and onboard systems requirements.

The cost of these items was then based on typical dollar/pound data.

(5) Cockpit Flight Controls

The flight control costs were based upon the requirement for pilot and co-pilot cyclic and collective controls and aft-pilot (cargo handling system operator) collective control and the pilot and co-pilot rudder pedals. These items are similar to those for helicopters.

(6) Instrumentation and Navigation Equipment

These requirements were considered similar to those for helicopters.

(7) Airconditioning and Anti-Icing Equipment

This equipment was based on expected control car requirements with the cost being estimated on a dollar/pound basis.

(8) Electronics Group

These requirements in part are similar to those for helicopters with the HLH type AFCS, PHS, and FBW electronics comprising the major additions. Data for these additions are based on available information from the HLH program which had similar requirements.

(9) Furnishings and Equipment

These requirements are similar to those for helicopters.

(10) Cargo Handling System

It has been assumed that a cargo handling system will be needed in certain HLA missions. The cost for this item has been based on an extrapolation of existing helicopter cargo handling system data.

(11) Assembly, Erection, Check-Out, Certification

The cost of this effort has been developed based upon recent corporate estimates and historical data relative to the erection of the envelope.

9.2.2 Development Costs

The development plan of Figure 10.1 has been considered representative of the type of development that the HLA configuration would receive. It is estimated that the development program of Figure 10.1 including FAA certification will require $\$50 \times 10^6$ given the GFE assumptions of Section 10 which includes four helicopters valued at approximately $\$24 \times 10^6$ in current dollars.

9.2.3 Total Aircraft Cost

The above development costs, as indicated in Figure 9.1, have been amortized over the quantity of vehicles produced and added to the flyaway cost to obtain the total aircraft cost. Also included in the total aircraft cost as a function of the quantity produced are:

(1) Facilities in which to assemble and erect the vehicle.

(2) Ground equipment required to handle the vehicle at the point of manufacture.

(3) Production tooling

A 90% learning curve has been conservatively applied in developing production quantity costs.

Table 9.2 provides a summary of the total aircraft cost versus the quantity produced.

9.2.4 Insurance Costs

Annual insurance costs were taken at 4% of the total aircraft costs which is twice the ATA rate for HTA and one-half that for helicopters.

It is generally believed that the insurance rate for the HLA will be much less than that typical for helicopters because

TABLE 9.2 SUMMARY OF TOTAL AIRCRAFT COST (MILLIONS OF DOLLARS) VERSUS QUANTITY PRODUCED[1]

	Quantity Produced			
	1	20	50	200
I Vehicle RDT&E Costs[2]				
Flight Research Vehicle (FRV)	18.50	18.50	18.50	18.50
Proof of Concept/Research Flight Test Program	2.50	2.50	2.50	2.50
Technology Programs	10	10	10	10
Modification to FRV	14	14	14	14
Flight Test and Certification	5	5	5	5
SUB TOTAL	50	50	50	50
II Operational Vehicle Costs[3]				
Envelope Group	1.84	26.09	50.88	164.90
Interconnecting Structure	3.15	44 66	87.10	282.30
Rotor/Turbine Module & Related Subsystems	19.6	199.90	441	1429
Control Car	0.22	3.1[illegible]	6.08	19.72
Cockpit Flight Controls	0.027	0.38	0.75	2.42
Instrumentation & Navigation Equipment	0.077	1.09	2.13	6.90
Airconditioning & Anti-Icing Equipment	0.012	0.17	0.33	1.08
Electronics Group	0.48	6.81	13.27	43.02
Furnishings & Equipment	0.024	0.34	0.66	2.15
Cargo Handling System	0.18	2.55	4.98	16.13
Assembly & Erection	0.15	2.13	4.[illegible]5	13.44
Helium	0.12	1.7	3.32	10.75
Supporting Engineering	0.08	1.13	2.21	7.17
Acceptance Flights	0.04	0.57	1.11	3.58
SUB TOTAL	26	290.64	617.97	2002.56
III Manufacturing Facilities		15	35	140
IV Aerospace Ground Equipment		1.25	1.87	5.62
V TOTAL AIRCRAFT COST (SUM OF ITEMS I, II, III & IV/QUANTITY)		17.84	14.10	10.99

[1] C&A and fee included, each at 10%

[2] See Section 10.0 for development program details

[3] 90% learning curve. Applied to each rotor/turbine module

of the inherent safety that the buoyancy provides from the standpoint of multi-engine failure.

9.2.5 Depreciation

The standard ATA approach for depreciation was used with respect to spares allocation with a straight line to zero residual assumed. The period of depreciation for the entire vehicle has conservatively been assumed to be 12 years which is typical of helicopter practice.

9.2.6 Direct Maintenance

Direct maintenance labor and material costs for the hull and LTA components were based on past airship experience. For an annual utilization of 1000 hours, two riggers and one mechanic will be required to maintain the LTA components of the vehicle. For an annual utilization rate of 2000 hours three riggers and two mechanics will be required. An annual salary of $14,000 (2000 man hours) has been considered for riggers and mechanics.

The direct maintenance costs for the interconnecting structure were based on the ATA approach for aircraft fuselages with four hours flight cycle considered as typical. A labor rate of $6.90/hour has been assumed which is the 1967 ATA rate of $4.00/hour escalated by the Consumer Price Index (CPI). Maintenance material dollars were also escalated by the CPI.

As noted previously, the maintenance costs for the rotor/turbine module and related subsystems was based upon data for the commercial operation of the S64F helicopter. For an annual utilization rate of 1000 hours, one chief mechanic and eight mechanics will suffice to maintain the rotor/turbine modules and related subsystems. For an annual utilization of 2000 hours, one chief mechanic and 10 mechanics will be required. An annual salary of $18,000 and $14,000 have been considered for the chief mechanic and mechanics, respectively.

Material costs for the rotor/turbine modules and related subsystems are based directly upon the requirements for the S64F which include the following:

(1) Repairables plus miscellaneous at \$126/hour

(2) Component overhaul at \$463/hour

(3) Engine overhaul (2 engines) at \$80,000 each at 800 hours MTBF plus 5% for parts at \$210/hour. Total \$799/hour.

The requirement for the HLA in this area is \$2876/hour which is four times the above total excluding the components deleted from Item 2 due to difference in configuration.

9.2.7 Maintenance Burden

Maintenance burden was considered at 30% of the direct maintenance labor.

9.2.8 Flight Crew Costs

The following flight crew and crew costs have been considered as a function of utilization rate:

	Utilization					
	1000 hrs		1500 hrs		2000 hrs	
	No. Reqd	Annual Cost	No. Reqd	Annual Cost	No. Reqd	Annual Cost
Pilots	2	\$50,000	3	\$75,000	3	\$75,000
Co-Pilots	1	\$21,000	2	\$42,000	2	\$42,000
Winch Operator	1	\$18,000	1	\$18,000	2	\$36,000
TOTAL		\$89,000		\$135,000		\$153,000

The above costs per crew member are typical of those experienced in the operation of the S64F helicopter.

9.2.9 Fuel and Oil Costs

Fuel and oil requirements were based on the power required at 60 knots from Figure 7.2 and the fuel consumption data of Reference 13. At the design gross weight of 147,365 kg (324,950 lbs) the estimated fuel consumption is $0.94 \times 10^{-9} m^3/kg\text{-}m$ (0.3[illegible] gal/ton-mile). Fuel costs were considered to be \$0.50/gal with oil costs considered to be 5% of fuel costs.

9.3 Utilization Rate

Utilization rate of the HLA can be expected to vary dependent on mission parameters. The crane helicopter itself in a commercial operation has been used at a 2000 hour/annum rate with 1500-1800 hours/annum more typical. Other large helicopters have been used commercially in excess of 2000 hours/annum.

Historically large airships have experienced actual utilization rates of 3'00 hours for a portion of a year (Reference 14). When projected to a total year, the resulting utilization rate is on the order of 5200 hours per annum. These operations were over long stage lengths and would not be realizable at the shorter trip distances the HLA will typically experience. Projected utilization rates, based on historical LTA data for stage lengths typical of those of the HLA, are in the 1500 hours/annum category. Although past LTA utilization experience is not directly applicable to the HLA, it does provide some indication that the LTA aspects of the vehicle will not prove the limiting element in annual utilization.

It is anticipated that dependent upon mission parameters that the utilization rate can be expected to be as large as 2000 hours/annum. However, since utilization rate will be a variable over the realm of possible heavy lift missions, the sensitivity to utilization rate in terms of TOC is presented subsequently in this economic analysis.

9.4 Indirect Operating Cost Analysis

The indirect operating cost elements listed in Table 9.1 were considered representative of those required to support a commercial operation. Such an operation might be wholly owned by a large industry requiring heavy lift services on a continuing basis. Several smaller industries with less frequent heavy lift requriements might structure a joint venture for operating a heavy lift vehicle. It is also considered possible that under certain circumstances HLA services may be provided on a rental basis to a series of users similar to current helicopter rental services.

9.5 Total Operating Costs

The DOC and IOC elements were combined to obtain the TOC. Table 9.3 summarizes the elements of the hourly TOC for the HLA flying at design gross weight at the design range at the design velocity as a function of annual utilization rate and production lot size. The four dominant elements in the TOC are in order of decreasing importance:

(1) Maintenance of the rotors, turbines, and associated controls

(2) Vehicle depreciation

(3) Fuel and oil costs

(4) Vehicle insurance costs

The most significant element of the above four elements is the maintenance of the rotors, turbines, and associated controls. Substantial confidence can be assigned to the magnitude of this

TABLE 9.3 HLA TOTAL OPERATING COSTS AT THE DESIGN GROSS WEIGHT AT DESIGN RANGE AT DESIGN VELOCITY

		PRODUCTION LOT SIZE								
		20			50			200		
		UTILIZATION (HRS/ANNUM)								
		1000	1500	2000	1000	1500	2000	1000	1500	2000
1.0	DOC ($/HR)									
1.1	FLYING OPERATIONS									
1.1.1	FLIGHT CREW	89	89	76	89	89	76	89	90	76
1.1.2	FUEL AND OIL	1040	1040	1040	1040	1040	1040	1040	1040	1040
1.1.3	VEHICLE, INSURANCE	711	474	356	562	375	281	438	292	219
1.2	DIRECT MAINTENANCE OF FLYING EQUIPMENT									
1.2.1	ENVELOPE	91	61	59	91	61	59	91	61	59
1.2.2	INTERCONNECTING STRUCTURE	47	42	39	47	42	39	47	42	39
1.2.3	ROTOR/TURBINE/CONTROLS	3004	2961	2954	3004	2961	2954	3004	2961	2954
1.3	DIRECT MAINTENANCE BURDEN	56	40	40	56	40	40	56	40	40
1.4	VEHICLE DEPRECIATION	1732	1154	866	1392	928	696	1092	728	546
2.0	IOC ($/HR)	151	101	76	151	101	76	151	101	76
3.0	TOC ($/HR)	6921	5962	5506	6432	5637	5261	6003	5355	5049

TOC element since it is based directly on commercial experience with similar helicopter components. It is believed that the magnitude of this element can be substantially reduced by implementing low maintenance components into the HLA design. The HLA concept permits larger design margins to be considered in the high maintenance dynamic components and accordingly it may be possible to significantly reduce this TOC element. The development plan of Section 10 of this report recommends further exploration of this possibility.

Substantial confidence can be attached to the fuel and oil costs since they are derived from actual fuel consumption data for existing helicopters. The fuel consumption in turn is based on power required curves which are quantifiable with reasonable accuracy at this point.

Less confidence can be assigned to vehicle insurance costs and vehicle depreciation costs because they are based on vehicle acquisition cost which can early in the study of a concept be the subject of considerable discussion. In an attempt to avoid some of this discussion a generally conservative approach has been attempted in the development of the initial vehicle cost and in the application of the learning curve effect.

9.6 TOC Comparisons

Comparison of the HLA TOC with that of the S64F helicopter is made in Figures 9.2 and 9.3. The TOC comparison on an available payload ton-mile (statute) basis is provided in Figure 9.2 as a function of annual utilization and quantity of vehicles produced. Both vehicles are considered to be performing a mission that the helicopter is capable of performing in terms of payload weight. Fighre 9.3 provides a comparison of TOC's as a function of range where the range is extended by use of available auxiliary external tanks. The payload of both vehicles are diminished to accommodate the increase in fuel required to achieve the range increase.

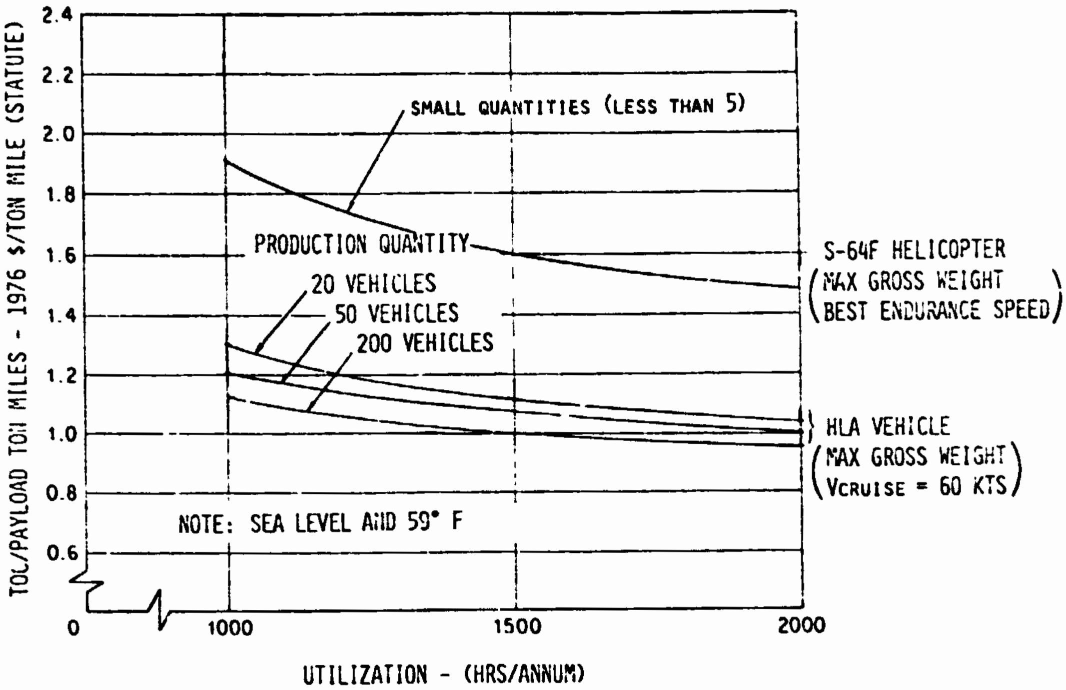

Figure 9.2 Comparison of S-64F and HLA Total Operating Cost/ Payload Ton Miles at Design Range of HLA

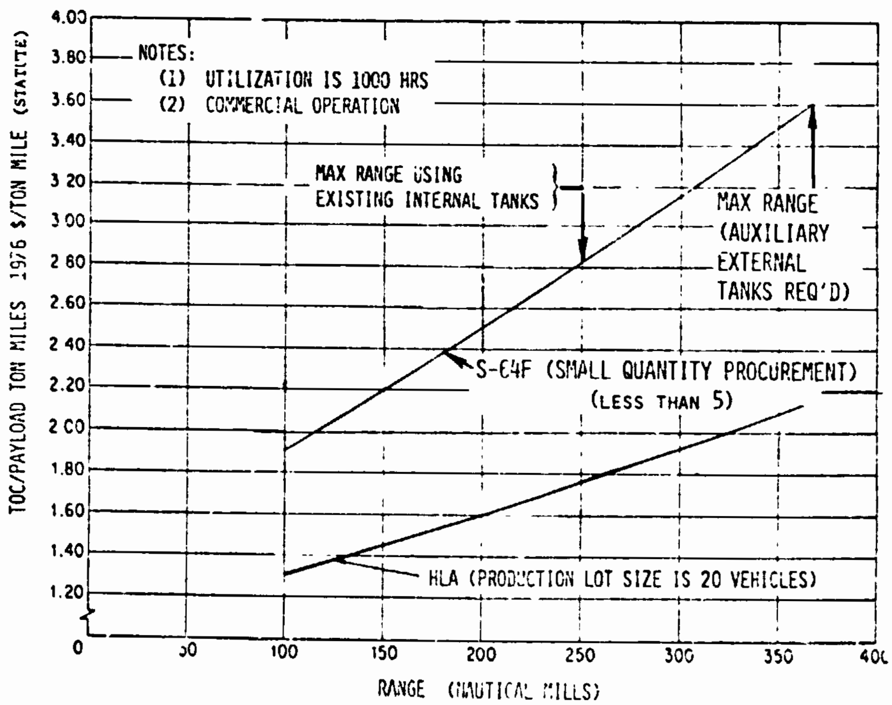

Figure 9.3 Comparison of S64F and HLA Total Operating Cost/Payload Ton Mile

These comparisons indicate the general economic benefits to be derived from combining of rotor and buoyant lift. The additional economic benefits that a vehicle capable of lifting large outsized loads can provide in terms of factory versus remote site assembly, special highway or roadways receiving only limited use, etc., have not been evaluated.

10.0 TECHNOLOGY ASSESSMENT ANALYSIS

10.1 General

A significant requirement of the current program was the Technology Assessment Analysis (TAA). The TAA included the following major sub-tasks:

(1) Identification of important technology areas where substantial contributions toward safety, economics, or performance could be achieved.

(2) Identification of the need for flight research vehicles.

(3) Identification of development costs and schedules.

The TAA has indicated that successful technology programs will contribute significantly toward improved economics, safety, and performance of the size of vehicle investigated during the Phase II study and toward larger vehicles that are projected for future civil and military needs. A primary example of a technology program which leads to economic benefit is the development and application of low maintenance rotor/turbine concepts. As illustrated previously in this report, the projected maintenance costs of the Phase II HLA are dominated by current maintenance requirements of the helicopters. The availability of buoyant lift to offset the weight of the rotor system components can permit greater margins on the dynamic components currently requiring a high degree of maintenance. Other recommended technology programs are discussed subsequently in this section.

The TAA has also indicated the need for a Flight Research Vehicle (FRV). For reasons developed subsequently in this section, it is recommended that a 75-ton configuration be considered as the point of departure during the FRV program. Goodyear's Phase II technical effor as reported in previous sections of this report have indicated that through proper application of existing technology and engineering techniques such a FRV can be constructed with a minimum of development. An FRV is required to secure basic concept verification and to obtain research capabilities that cannot be duplicated in ground-based facilities or in ground-based component and subsystem testing. An additional dimension that will be provided is that a means will be available to illustrate advances afforded by the new technology emanating from the recommended technology programs as well as other rotorcraft and general aeronautical systems advancements. The research capabilities of the FRV coupled with the full-scale evaluation of new technologies will result in an adequate technical base permitting the development of larger HLA vehicles meeting projected civil and military requirements.

As noted previously in this report, projected civil needs include payload capacities of several hundred tons while current military requirements range up to 140 tons.

In summary, the TAA has resulted in the overall development plan shown in Figure 10.1. The major features of this plan include the FRV development program (Item 1 of Figure 10.1) in parallel with the recommended technology progams (Item 3 of Figure 10.1). The basic FRV once evaluated would be modified (Item 4 of Figure 10.1) to evaluate the promising new technologies emerging from the recommended technology programs. The details of the FRV development program and the recommended parallel technology programs are discussed in more detail subsequently in this section.

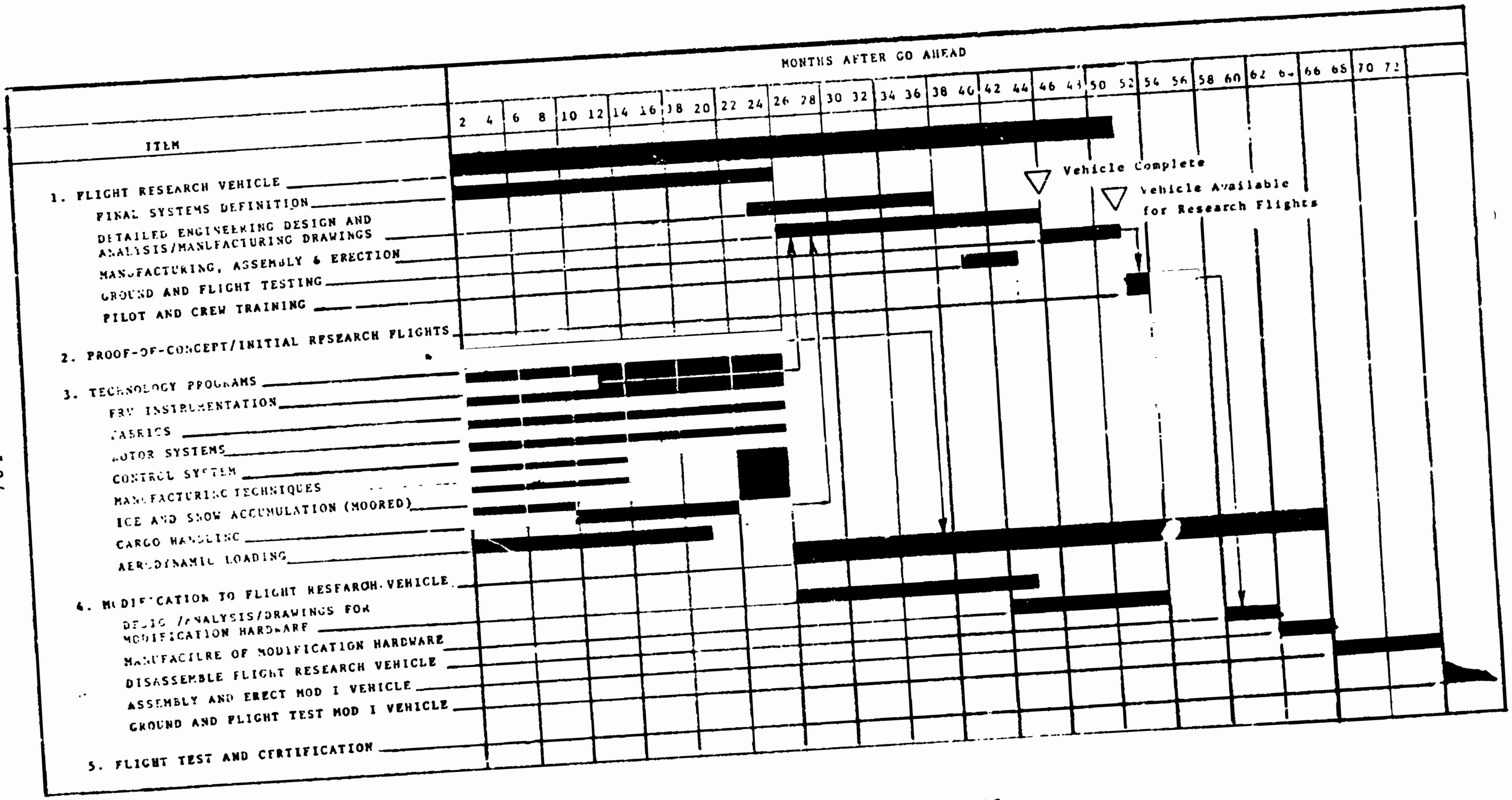

Figure 10.1 Heavy Lift Airship Development Plan

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10.2 Technical, Economic and Institutional Considerations Requiring Full-Scale Data From Flight Research Vehicle

10.2.1 General

Major areas requiring flight research vehicle data prior to achieving an adequate technology base permitting the development of larger civil and military vehicles are included in Figure 10.2. Also illustrated are the type of economic and institutional considerations requiring full-scale data prior to user and regulatory agency acceptance of the HLA class of vehicles. Potential users (military and civil) will rely heavily upon the actual operating cost, availability, and utilization data generated during the recommended FRV economic experiments in their acceptance of this new class of vehicles. Potential users will be interested in a clear demonstration that the prior airship deficiencies of low speed control, ground handling, and ballast/load interchange have been either eliminated or greatly minimized.

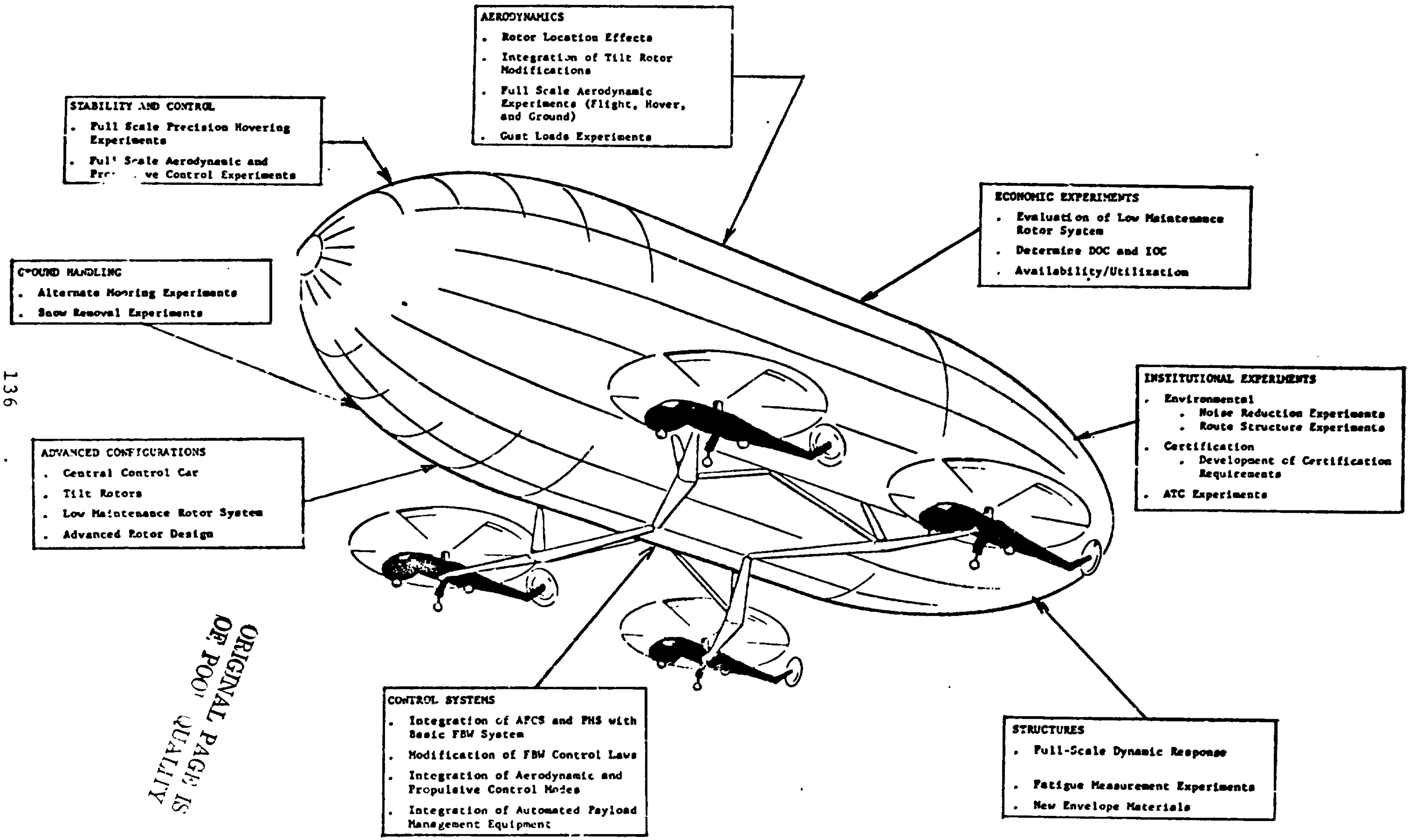
Government regulatory agencies will rely upon the FRV for the needed verification that the design criteria, operational procedures, and ATC approaches are adequate to permit this new type of vehicle to safely enter the national transportation picture. The FRV will also permit the development of criteria which must be met by a potential user in filing flight plans, obtaining route clearnace, etc.

Detailed comments on all elements of the plan depicted in Figure 10.2 are not presented here; however, the following are typical justification of the need for full-scale data.

10.2.2 Requirement for Full-Scale Aerodynamic Data

Analysts of the aerodynamic characteristics of airships have long realized the benefits and limitations of scaled testing in even the largest wind tunnel facilities. Fortieth scale tests of the Akron configuration were conducted in the mid-1930's with more recent large-scale model tests (late 1950's) involving the testing of a similar sized model of a non-rigid configuration in the Langley 30 x 60-foot facility. While it is well known that

FIGURE 10.2 TECHNICAL, ECONOMIC, AND INSTITUTIONAL CONSIDERATIONS REQUIRING FULL-SCALE DATA



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wind tunnel evaluations normally serve a very essential role in the development of any new airship configuration or a change in a prior configuration, it is also recognized that full-scale flight testing is the only method of securing data uncompromised by scale effects. The wind tunnel testing of helicopter rotor systems has advanced to quite a sophisticated level. It is well known, however, that rotors are very sensitive to the dynamic interactions between the flexible rotor system and the unsteady aerodynamic environment in which it operates. As a result, full-scale flight tests of new rotor systems are considered mandatory. This is one factor that led to the NASA/Army program to develop the Rotor Systems Reseach Aircraft.

Initial exploratory aerodynamic evaluation is often accomplished with modest scale models in tunnels that permit quick changes in model configuration and test condition. Vehicle performance as it is effect by configurational changes is examined in this manner to support development and refinement of original vehicle concepts or configurations. Valuable insight into feasibility can often be secured on a cost effective basis through the use of modest scale models. Wind tunnel data in general are often essential in the development of aerodynamic theory and methods of analysis relative to new vehicle configurations. Of course, final design data for an initial flight vehicle is best secured from testing at the largest possib[illegible] compatible with existing facilities.

The development program discussed subseque[illegible] in this section utilizes the above methodology in se[illegible] [illegible]equate aerodynamic data to design the FRV. However, it m[illegible] be recognized that the HLA configuration with its combining of large rotors in close proximity to a large hull certainly cannot be completely characterized aerodynamically in existing wind tunnel facilities. As a result, in order to secure adequate aerodynamic knowledge to ultimately permit the development of larger civil and military vehicles of this type, a flight research vehicle is essential.

10.2.3 Requirement for Full-Scale Control System Data

The control system for the HLA is unique with respect to past airships in that a FBW control system directs the appropriate combination of thrust vectors to achieve the desired vehicle response whereas prior airships have relied mainly on aerodynamic control surfaces. Computer modeling supported by input data from wind tunnel testing will continue to be utilized during the FRV development program to describe vehicle hovering and flight control capabilities. Computer modeling supported by inputs from the wind tunnel testing of a structural dynamics model will further assist in the synthesis of control laws and control system requirements wherein the aerodynamic, structural, and control system interactions are accounted for. It is also anticipated that existing ground-based simulators will be of significant use in the FRV development and pilot training once a sufficiently representative flight dynamics model is available. However, only via actual performance evaluations with the FRV can the adequacy of these design tools be evaluated. It can certainly be expected that the FRV itself will serve as the final design tool with respect to the ultimate definition of the control system for this type of vehicle. The FRV will offer the capability to perform various flight and hover mode operations either manually or automatically. Thus a means will be available to define exactly what automatic modes are essential and what modes are best left to direct pilot control. The FRV control system will also offer the capability to adjust critical sensitivities such that the system response can be appropriately altered based on flight test program results. Control laws will also be reasonably easy to modify or increase in sophistication as may be dictated by flight test program results.

10.3 Research Vehicle Instrumentation

Successful flight research is greatly dependent upon the ability to accurately measure and record the appropriate data from which to calculate the states and parameters defining the characteristics of interest. All functions of interest must be instrumented with sensors and signal conditioning compatible with the data recording system. The system selected for consideration on the HLA is the Piloted Aircraft Data System (PADS), a new and versatile data collection system designed at Langley Research Center specifically for aeronautical flight research programs including rotorcraft.* A PADS provides up to 104 PCM channels for recording data up to 10 Hz, and up to 40 constant bandwidth FM channels for use in recording up to 400 Hz data. In addition, one channel is provided for recording voice and events and one channel for recording PCM time code for use in correlating measurements rec[illegible]ed onboard and measurements telemetered to the ground station.

In general all data would be recorded using inflight magnetic tape by an onboard instrumentation engineer in the command helicopter. As required, additional PADS would be employed at one or more slave helicopters which would be operated by additional instrumentation engineers.

The instrumentation system in addition to measuring the basic flight data such as attitude, airspeed, altitude and angle of attack [illegible]ld also be used to measure the type of data listed in Table 10.1.

It should be noted that the definition of instrumentation requirements and system for the HLA does not require any new technology. It does, however, require considerable adaptation of existing technology and techniques. As a result, an independent entry has bee[illegible] included in the list of recommended technology programs to develop the [illegible]RV instrumentation requirements and system hardware.

10.4 Need and Cost Benefit of Serial Development

One item specifically requested in the SOW was to explore the need or cost benefit of serial development relative to flight research vehicles. As noted previously, the Phase II 75-ton payload vehicle has been recommended as a point of departure for the

*The PADS is currently used in the Rotor Systems Research Aircraft under development by Sikorsky for NASA/Army

TABLE 10.1 LIST OF FRV DATA REQUIREMENTS

Vehicle Component	Type of Measurement	Conditions of Interest
Helicopter	Command and Slave rotor/turbine variables Load inputs to interconnecting structure	Flight and Hover (IGE & OGE)
Control System	Position of command pilot sticks and pedals Fly-By-Wire commands to slave and command helicopters	Flight and Hover (IGE & OGE)
Envelope	External pressure distribution Fabric loads Internal pressure	Flight and Hover (IGE & OGE); Moored
Suspension System	Cable loads Fabric loads	Same as envelope
Interconnecting Structure	Distribution of loads	Same as envelope
Vehicle	Pressure Rakes Landing gear	Same as envelope Landing & Moored
Cargo Handling System	Payload dynamics Sling Loads	Flight and Hover (IGE & OGE)

recommended FRV program. With respect to the recommended vehicle, there appears to be no technological need that requires the development of a smaller vehicle as a prerequisite. Present appraisals indicate that a small vehicle that addresses the real technical issues associated with the large concept would not appreciably alter the technical risk of the program. As noted in Section 10.5, a more meaningful assessment of serial development benefits will be possible after the 24-month Final System Definition Phase of the recommended FRV program. It is clear that a smaller or less elaborate initial FRV or a combination of both could lead to reduced costs of the initial program given the same GFE assumptions as considered in the recommended program. It is likely in the final analysis, however, that a more modest intial approach will not result in lower overall costs to develop a technology capability sufficient to provide vehicles meeting the projected civil and military requirements including vehicles larger than 75 tons.

Goodyear has explored the cost benefits associated with smaller initial vehicles including consideration of the ZPG-2W envelope currently in storage at NAF Lakehurst. As a part of the Phase II TAA, a point design study was performed during which the ZPG-2W envelope was examined for potential use in the initial HLA vehicle. The general arrangement of such a vehicle is illustrated in Goodyear Drawing 76-321 (see Figure 10.3 of this report). Furthermore, in attempting to define a minimum cost vehicle only the basic FBW control system was considered, thus inclusion of the HLH AFCS and PHS were excluded in this vehicle.

The helicopters in the configuration shown in Drawing 76-321 are gimbaled in pitch only and do not have the aft tail rotors reoriented. Both of these simplifications were considered in an attempt to reduce overall vehicle complexity and cost. Such a configuration would result in the following approximate capabilities:

(1) 40-ton payload

(2) Non-refueled range of 60 NM

(3) Maximum forward speed: 60 kts (TAS)

In the development of a plausible ZPG-2W HLA configuration, a very simplistic initial configuration was considered in order to minimize alterations to the existing ZPG-2W envelope. This initial configuration retained the existing internal suspension system and added only an external suspension system which interfaced at the envelope equator in order to obtain maximum control over the envelope in response to gust and rotor induced loads. The interconnecting structure was maintained external to the envelope again in an attempt to minimize complexity and cost. Unfortunately, the resulting configuration was only able to resist approximately one sixth of the yawing moment that the helicopters can create.

In order to increase the yawing moment capability the general arrangement shown in Goodyear Drawing 76-321 was adopted.

This permits a wider internal system, similar in concept to the Phase II configuration, to be added which increases the ability to resist helicopter yawing moments from one-sixth to one-third that which is available. Further analysis of the ZPG-2W HLA configuration was discontinued realizing that significantly increased yawing moment capability could not be achieved and that the capability which was achieved would suffice at least for demonstration of flight purposes. Control system limiters would, however, be necessary to prevent collapse of the envelope.

Major modification is required to the bottom side of the envelope to achieve the configuration shown in Drawing 76-321. In addition, the use of the existing ZPG-2W envelope requires replacement of an area of defective fabric on the top side of the envelope.

It is clear from the 2W HLA design study that a vehicle less expensive than the Phase II configuration can be developed. Such a vehicle in general will have compromised proof-of-concept and research capabilities. The configuration involving the use of the ZPG-2W is considered to be a less than "minimally acceptable configuration" from either a proof-of-concept or flight research vehicle standpoint. There is obvious middle ground between the ZPG-2W HLA and the Phase II HLA configuration that could result in an acceptable FRV approach should future funding situations dictate such an approach. Without knowledge of the precise nature of such a potential situation, Goodyear recommends that as a point of departure, the present Phase II HLA configuration and the research and proof-of-concept attributes attached thereto be considered for future development.

For maximum proof-of-concept and research capabilities an FRV should meet the following criteria:

(1) The vehicle's payload capacity should be sufficiently larger than current helicopter designs to attract sufficient initial interest and to provide a proof of the lift benefits claimed for the concept.

Figure 10.3 HLA Configuration Using ZPG-3W Envelope

(Goodyear Drawing 76-321)

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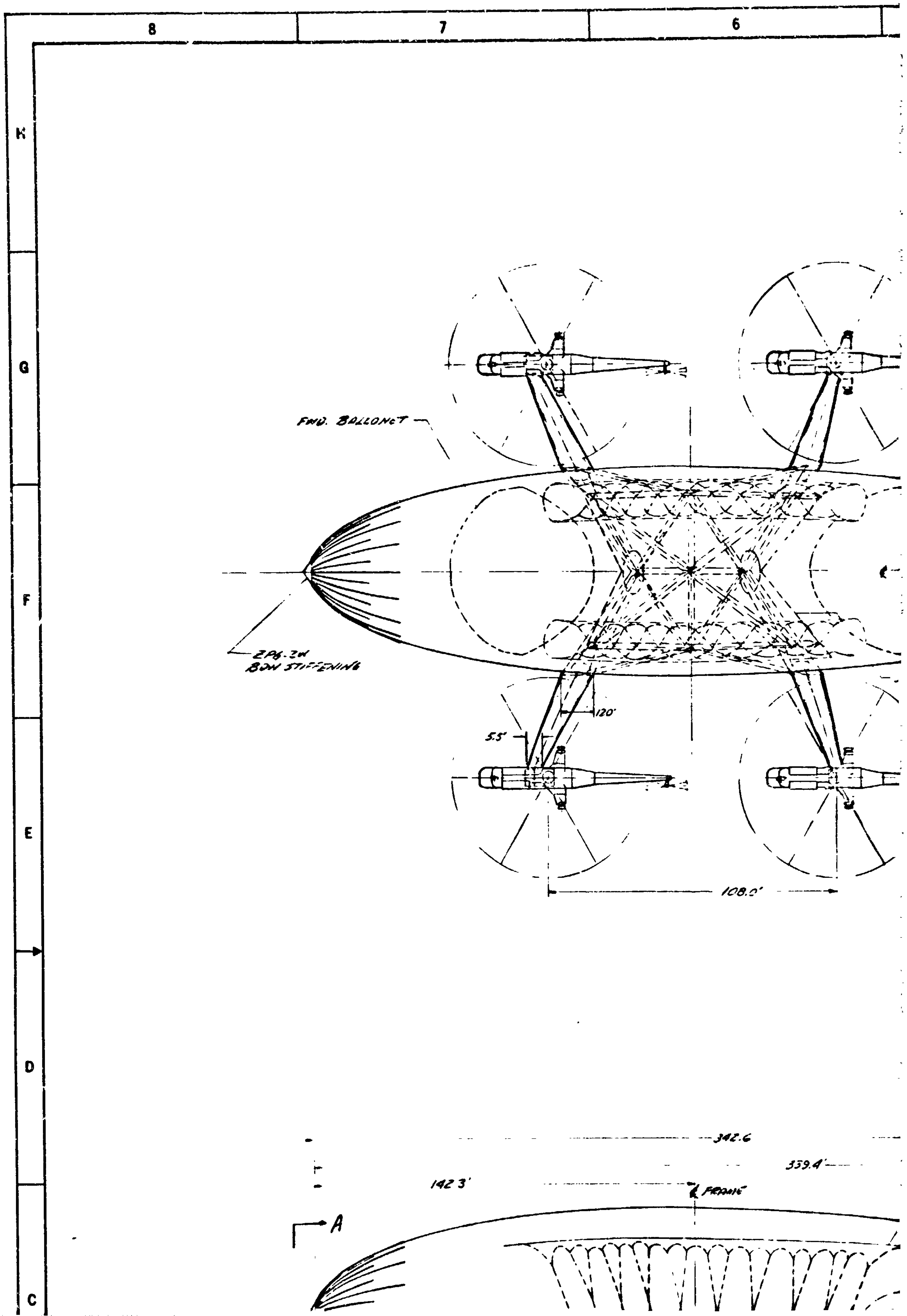

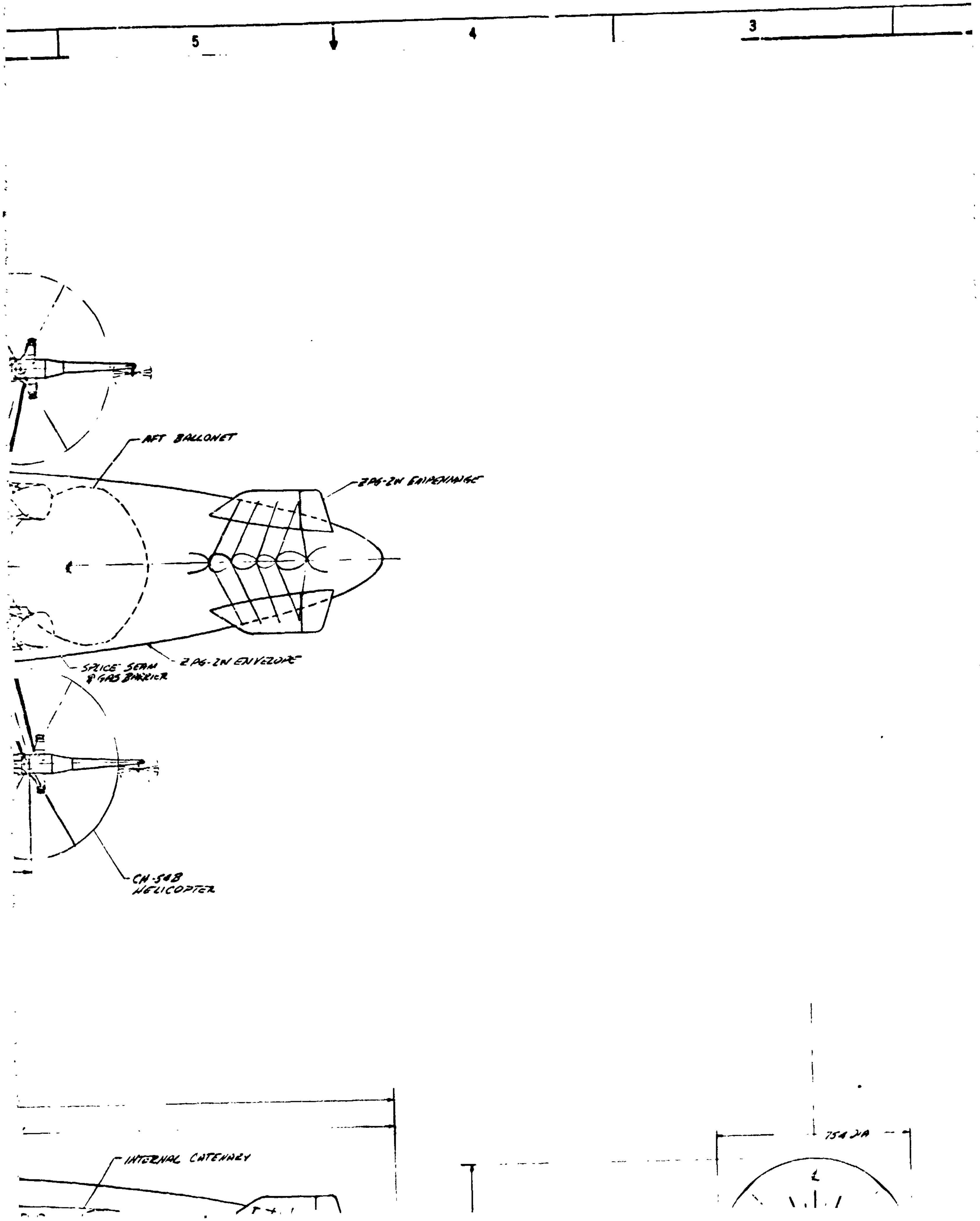
5
4
3
AFT BALLONET
ZPG-2W EMPENNAGE
ZPG-2W ENVELOPE
SPLICE SEAM & GAS BARRIER
CH-54B HELICOPTER
INTERNAL CATENARY

FOLDOUT FRAME 3

3	2	1

REVISIONS

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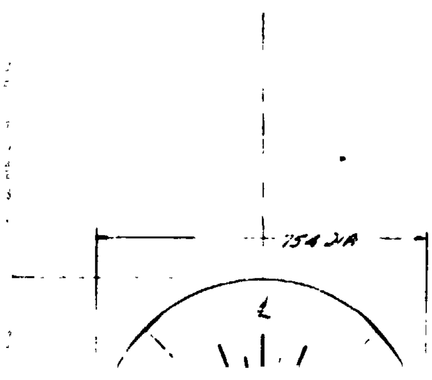

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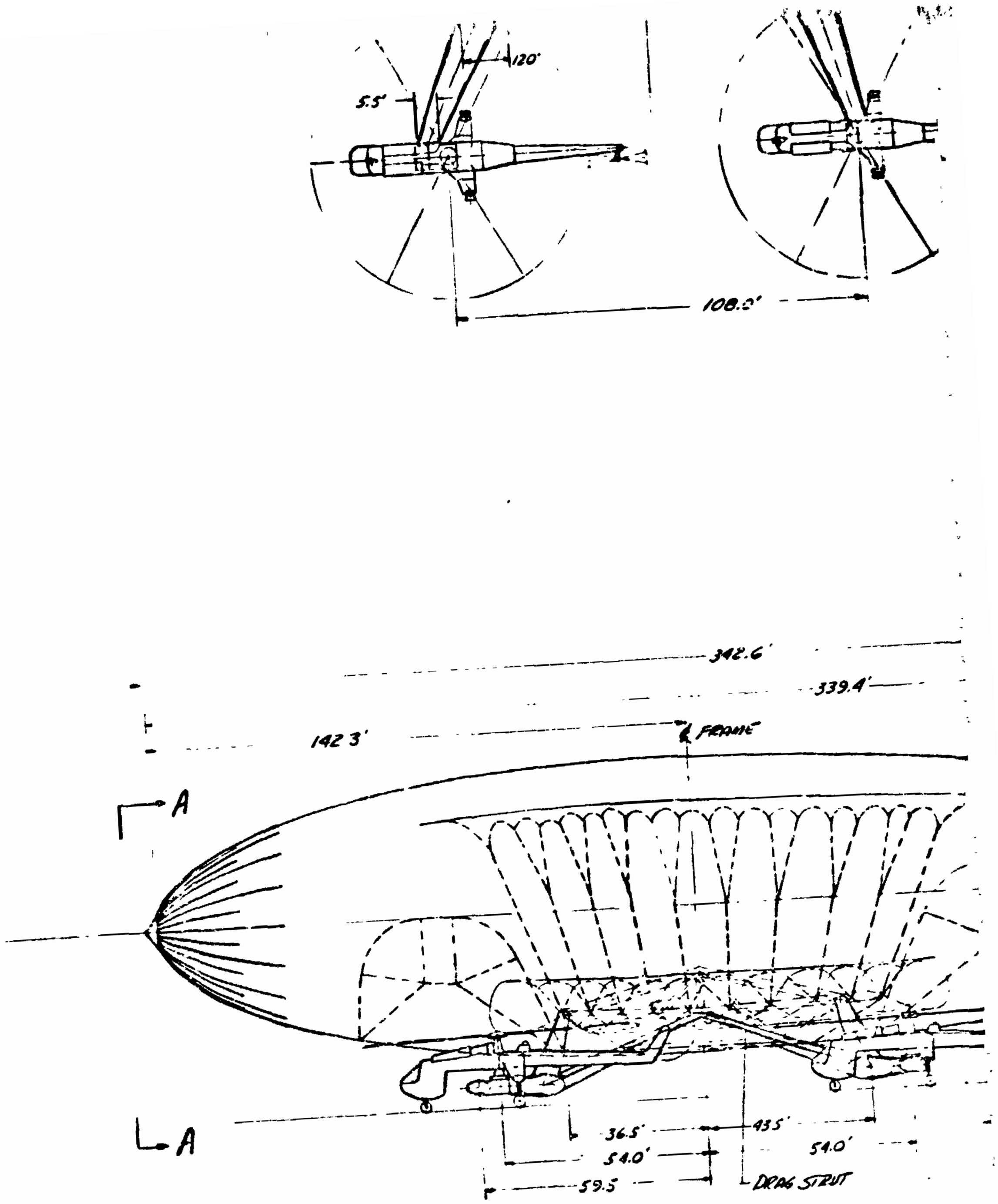

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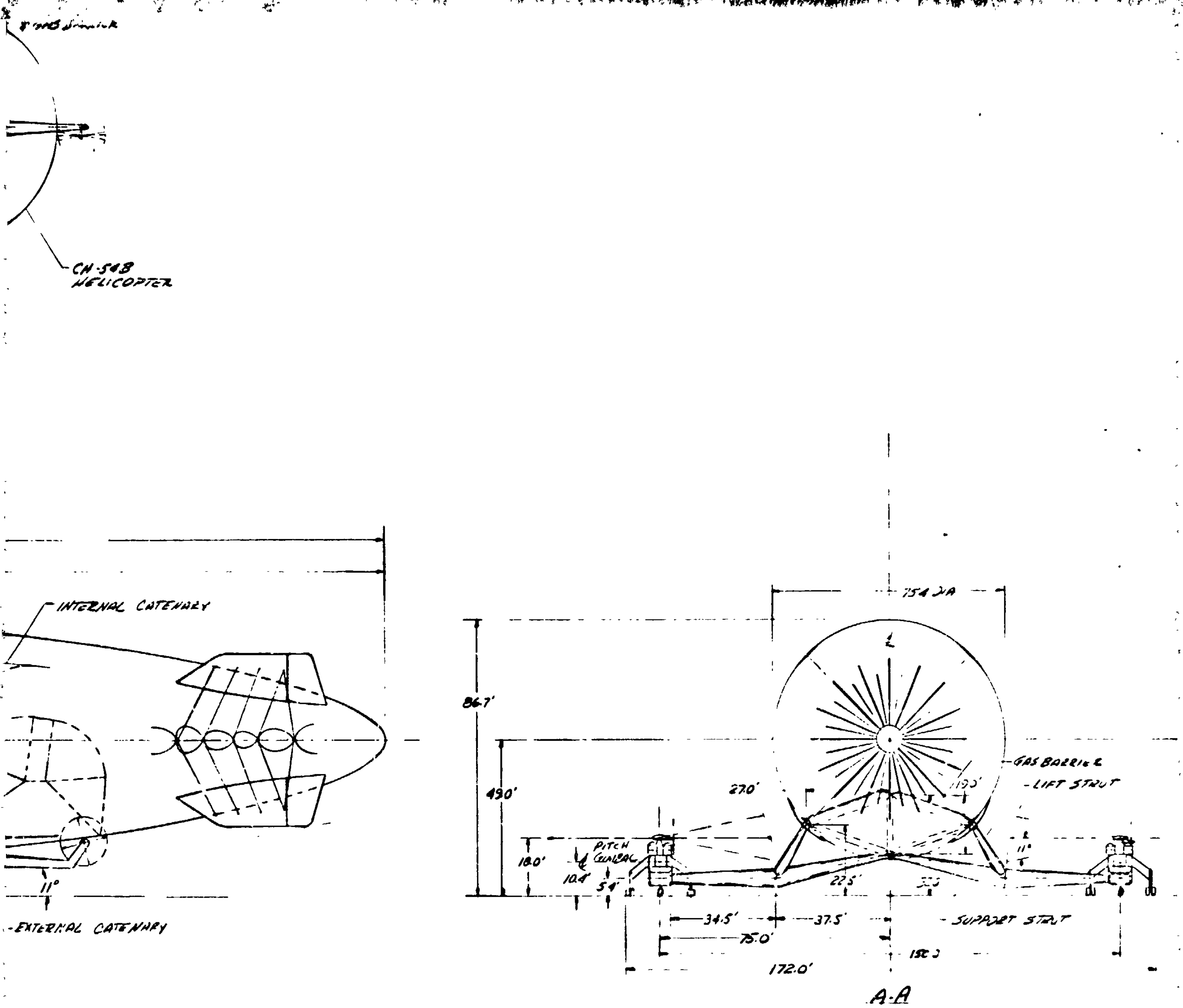

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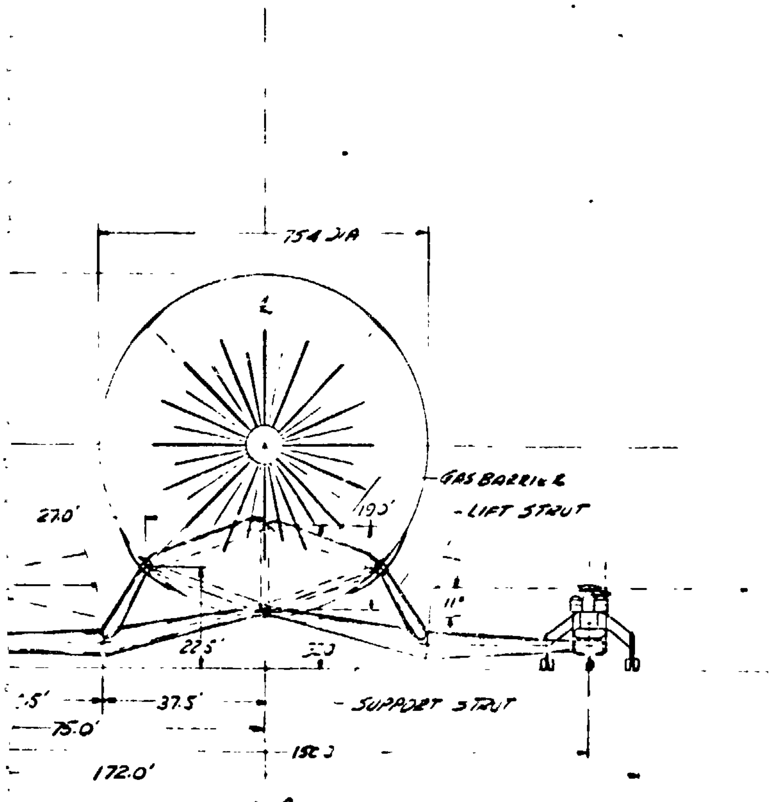
MATERIAL.

SEE GR60-9-60 FOR DRAWING INTERPRETATION STANDARDS

FOLDOUT FRAME 5



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FOLDOUT FRAME 6

(2) The vehicle should be of sufficient size to provide data relative to the operational complexities and operating costs associated with vehicles meeting current and projected heavy lift requirements. However, the size of the initial vehicle must be sufficiently compact as to fit within existing erection facilities.

(3) For any given helicopter selection, the propulsive thrust to maximum take-off gross weight ratio should be a minimum consistent with sufficient controllability. This condition insures a demonstration of the maximum payload capacity claimed for the HLA concept for any given helicopter.

(4) The AFCS of the helicopters selected should have a capability of accepting both electrical and conventional mechanical control inputs.

(5) The helicopters selected should have dual engines with a high percentage of reserve power such as not to significantly penalize the configuration due to the one engine out design requirement.

(6) The initial vehicle, assuming a multi-rotor system, necessitates a FBW control system if any significant research capability is to be achieved. In addition the option to evaluate automatic versus manual flight control modes appears essential from a research capabilities standpoint.

(7) A precision hovering sensor is necessary to prove the concept can achieve sufficient hovering precision in adverse weather to perform various civil and military missions.

The Phase II configuration successfully complies with these criteria and as a result is believed to represent a very significant advancement in securing a research capability sufficient to acquire a technology base adequate to develop operational vehicles meeting projected civil and military requirements.

10.5 Description of Recommended FRV Development Program

The recommended FRV development program is an attempt to identify and integrate those specific areas of design, analysis, simulation, test, etc., required to attain a minimum risk program aimed at developing a vehicle possessing the needed research capabilities.

Many of the FRV development program elements can be considered technology programs in themselves and could be funded as such if the entire FRV program cannot proceed in the near term. In that eventuality it is recommended that each discrete program still be directed at the same specific design point. In many cases these programs would involve the adaptation of existing technology, methods of analysis, methods of test, development of predictive theory, etc., commonly used in rotorcraft, airship and airplane development.

The FRV development program outlined in Figure 10.4 includes an initial 24-month Final System Definition (FSD) Phase. The output of the FSD Phase is a configuration to which a high degree of confidence can be assigned such that the follow-on phases can be pursued on a low risk, timely basis. The FSD Phase includes:

(1) Continued configurational exploration in the ARC 12-foot wind tunnel facility relative to rotor location, hull fineness ratio, and hull cross sectional shape. The model size, instrumentation requirements, etc. would be similar to the model used in recently completed HLA testing in the ARC 7 x 10-foot facility.*

(2) An evaluation of the final configuration in the ARC 40 x 80-foot wind tunnel facility. This model would be on the order of 22 feet in length and 7 feet in diameter and would include actual helicopter

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*These items are included under development testing effort of Figure 10.4.

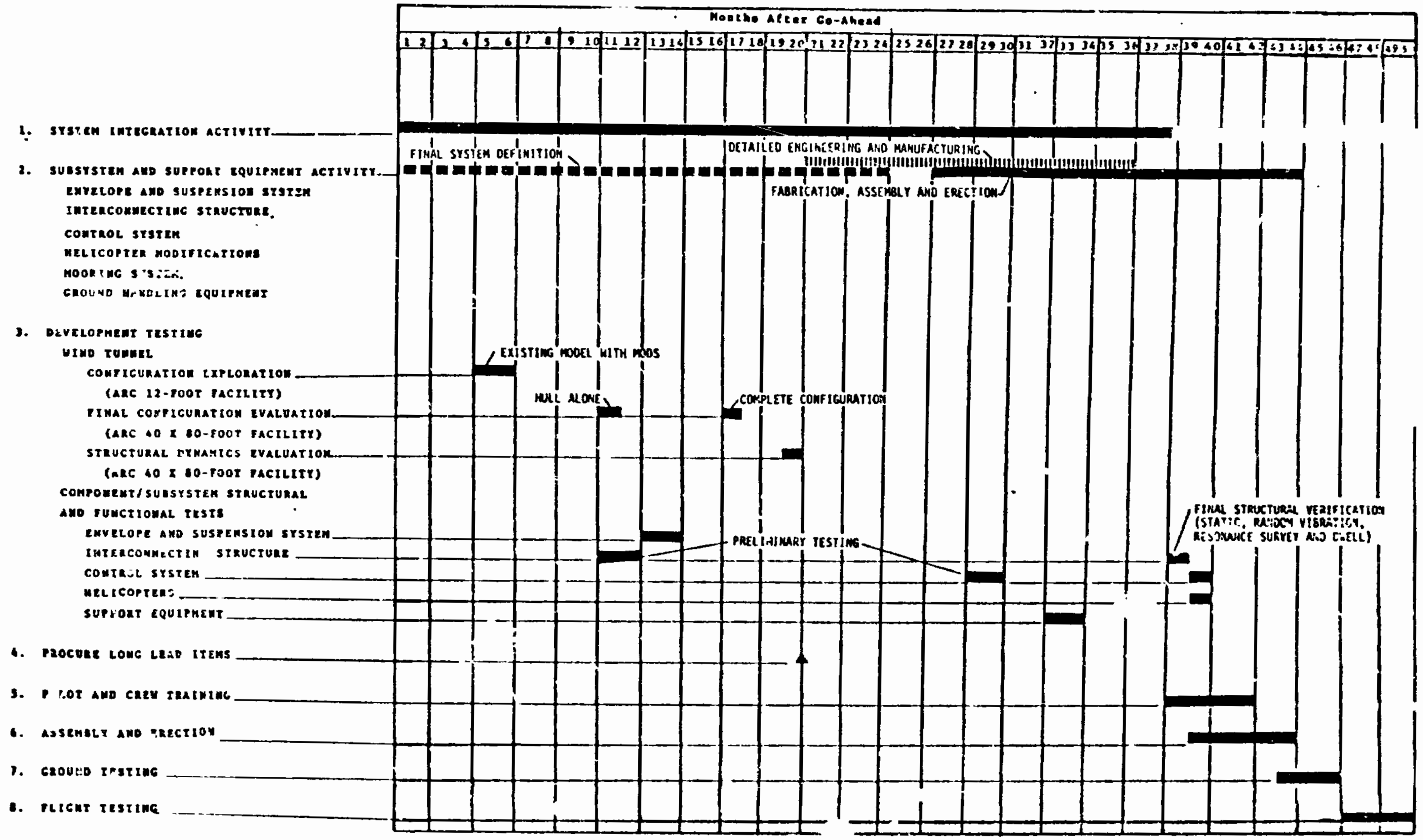

Figure 10.4 Flight Research Vehicle Development Program

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rotors with both collective and cyclic pitch capability. The rotor diameter would be on the order of four feet.*

(3) Additional consideration of a structural dynamics model in the ARC 40 x 80-foot wind tunnel facility.*

(4) Continued development of methods of analysis to explore:

a. Vehicle flight dynamics in hover, forward flight and when moored.

b. Theoretical aerodynamic predictive techniques

c. The overall vehicle structural dynamics resulting from the interaction of the vehicle structure, control system and unsteady aerodynamic environment.

d. Envelope and suspension system behavior under the various loading conditions associated with forward flight, hover and mooring.

(5) Development of final design criteria.

(6) Continued definition and refinement of current control laws and control concepts.

(7) Continued definition and refinement of the current mooring concept.

(8) Continued definition of the envelope/suspension system, interconnecting structure, helicopter/interconnecting structure interface, and helicopter modifications.

(9) Component and sub-assembly testing.

All of these elements have been integrated into the rudiments of a suggested methodology for the FSD Phase of the FRV program which is shown in Figure 10.5. These elements are all

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*These items are included under development testing effort of Figure 10.4.

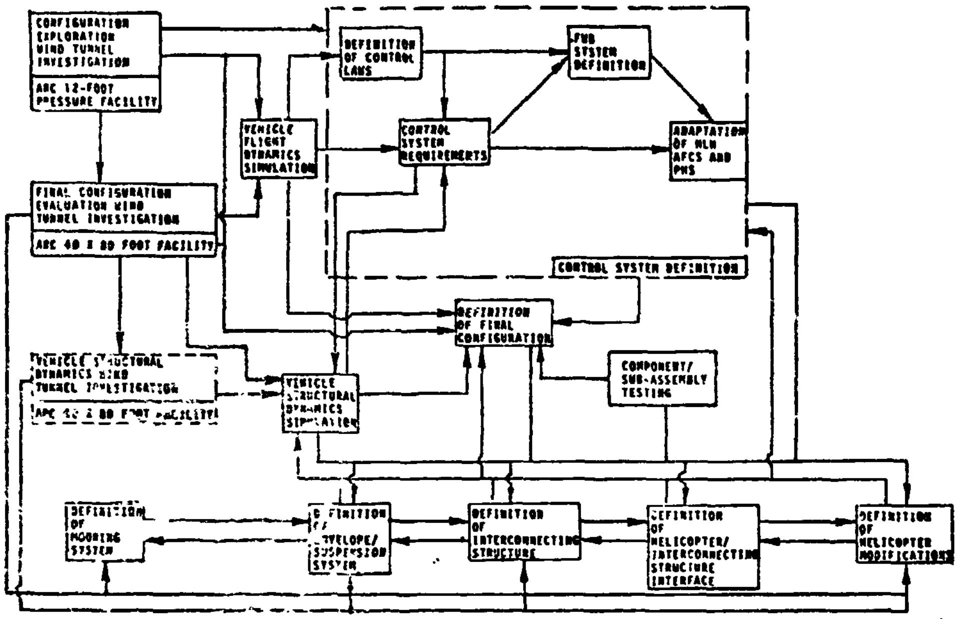


Figure 10.5 Methodology of Final System Definition Phase of FRV Program

focused towards the definition of the final configuration. It should be realized that the flows indicated in the figure would be continuous over the 24 months of the FSD Phase and that no attempt has been made to indicate the many iterations that will naturally occur. Also note that no attempt has been made to illustrate the general design and analysis activity at the sub-assembly or detail level or in fact the system integration design and analysis effort associated with the block entitled "Definition of Final Configuration."

It is recommended that the completion of the FSD Phase serve as a milestone for reviewing all aspects of the configuration and remaining elements of the FRV Program. Information may be available at that point that would suggest an interim configuration should be pursued, that the structure of the remaining program elements should be altered, etc.

The following paragraphs provide some insight into the details of certain of the elements of Figure 10.5.

10.5.1 Configurational Exploration in ARC 12-Foot Pressure Facility

The testing of the Goodyear HLA model in the ARC 7 x 10-foot facility has revealed several significant points in addition to confirming the basic feasibility of the concept. Significant indications of the continued value of configurational exploration include:

(1) The interference effects that were observed are a strong function of the rotor location with interference decreasing as the rotor is moved outboard.

(2) Favorable interference may be achievable.

(3) The modification of the flow field around the hull by the rotors may be a usable phenomenon in controlling the vehicle.

In general, Item (1) above suggests that sufficient additional rotor location testing is in order to minimize the interference between the rotors and the hull. Also suggested a a result of Item (1) above, is that higher fineness ratio hulls should be explored. The higher fineness ratio hull will have a similar affect to that of moving the rotors farther outboard for a given fineness ratio. The fineness ratio can be increased within reasonable limits without appreciable penalties being incurred. Finally, Item (1) above suggests that a different hull cross section may have a mitigating effect on the observed interference and it is believed that such configuration should be explored in any attempt to develop a flight research vehicle.

It is believed that by changes in vertical location of the rotor plane it may be possible to obtain a no-interference or even a favorable interference effect. The presence of such a region might result in a minimization of aero-elastic effects that may occur for a rotor system in an unsteady region.

During the testing of the Goodyear HLA model in the ARC 7 x 10-foot wind tunnel facility it was noted that by appropriate variation of upstream and downstream rotor thrust levels rather large hull forces could be generated which countered the cross-wind force on the hull. More testing is necessary to fully understand the possibility of using this approach to increase the crosswind hovering capability. If such an approach proves usable, the precision hover mode control laws would be altered accordingly.

It is necessary to move from the ARC 7 x 10-foot facility to the ARC 12-foot pressure facility because Reynolds Numbers much closer to the full-scale value can be obtained with the variable density feature in the 12-foot facility.

The output of the configuration exploration testing would be used in the following manner:

(1) To support continued development and refinement of the Goodyear vehicle flight dynamics computer simulation primarily in the development of a complete interference model.

(2) To support refinement of the overall arrangement of the HLA configuration. These refinements would generally be directed at improving the crosswind hovering capability of the current configuration.

(3) To support final definition of the configuration to be tested in the ARC 40 x 80-foot wind tunnel. In addition, the testing in the 12-foot facility will materially reduce the time required in the larger 40 x 80-foot facility.

(4) To support possible improvement in current vehicle control concepts and laws.

10.5.2 Final Configuration Evaluation in ARC 40 x 80-Foot Facility

It is recommended that the FSD Phase include a series of testing i the ARC 40 x 80-foot facility. The major purpose of this testing is to secure, short of full-scale, the best final design data obtainable. In general, the following type of data will be obtained in the 40 x 80-foot facility testing.

(1) Hull Pressure Distribution For:
  a. Moored (unpowered)
  b. Hover (IGE and OGE; powered and unpowered)
  c. Forward Flight (powered and unpowered)

(2) Static Force and Moment Coefficients (Unpowered)
  a. Moored
  b. Hover (IGE and OGE)
  c. Forward Flight

(3) Total Force and Moment Coefficients (Powered)
  a. Hover (IGE and OGE)
  b. Forward Flight

(4) Flow Visualization Studies

The data obtained will support the following FSD elements:

(1) Verification of overall configurational adequacy

(2) Finalization of aerodynamic inputs to the vehicle flight dynamics computer simulation.

(3) Fin lization of aerodynamic interference model ed in vehicle flight dynamics simulation.

(4) Finalization of aerodynamic inputs into computer analysis for predicting envelope and suspension system behavior.

(5) Definition of mooring system, interconnecting structure, and helicopter modifications.

10.5.3 Structural Dynamics Testing in ARC 40 x 80-Foot Facility

It is recommended that further consideration be given to the development of a structural dynamics model for test in the 40 x 80-foot facility. The structural dynamics model would include an inflatable fabric envelope with suspension system and a dynamically scaled interconnecting structure. The total feasibility and cost benefit of this suggestion has not been defined and is considerably beyond the scope of this portion of the Phase II Study. The model, however, would be oriented toward the development of an empirical technique for viewing the nature of the interaction of the rotor systems (operating in a potentially unsteady flow) with the envelope and interconnecting structure assembly. Methods for alleviating any undesirable structural dynamics that might be observed could be tried and evaluated. The model results could also serve as a method for checking the structural dynamics computer model.

10.5.4 Continued Development of Methods of Analysis

10.5.4.1 Vehicle Flight Dynamics Simulation

Another area of considerable activity during the FSD Phase of the FRV program includes the continued development and refinement of the current Goodyear 6 DOF vehicle flight dynamics simulation. The continued development and refinement of this simulation would include:

(1) An improved rotor math model for estimation of rotor stability derivatives over a wider range of rotor thrust levels.

(2) An improved gust model accounting for the interaction of atmospheric disturbances with the hull and individual rotors.

(3) An improved definition of the applicable turbulence spectrum over which the response of the vehicle should be analyzed.

(4) Inclusion of the helicopter kinematics when gimbaled.

(5) Inclusion of payload dynamics for those cases where the payload may necessarily have several degrees of freedom with respect to the vehicle.

(6) Inclusion of more realistic characteristics for the AFCS, PHS, and autopilots.

(7) Inclusion of control law options which may improve vehicle directional control as well as precision hover control.

(8) The aerodynamic inputs to the simulation would be continually upgraded as wind tunnel data became available.

In general the vehicle flight dynamics simulation in its various stages of development and refinement would be utilized to support the following:

(1) Synthesis of overall control system requirements and methodology.

(2) Direct synthesis of the fly-by-wire control laws and autopilots.

(3) Verification that the control laws developed for interface with the AFCS and PHS modes are compatible with manual modes.

(4) Synthesis of modifications to helicopter control system and tail rotors.

(5) Synthesis of methods of restraining payload for expected flight and hover environments.

(6) Definition of final configuration.

As noted earlier, once a sufficiently representative flight dynamics model is developed it is recommended that an existing ground based simulator at NASA-Ames be programmed for purposes of pilot evaluation and training.

10.5.4.2 Theoretical Aerodynamic Predicitive Techniques

The major objective of this task is to develop three-dimensional predicitive techniques for calculating hull pressure distributions and general flow field characteristics including interference effects for the flight conditions of interest. The method will use potential flow for modeling the actuator disc and the hull. Boundary-layer separation effects will be included because these have a large influence on the flow field of the downwind rotors. In addition, ground effect will be included using imaging schemes. The overall analysis will be computerized for rapid computation.

The major purpose in developing this analytical tool is that all variations of interest cannot be evaluated in the wind tunnel because of the complexity of testing at some conditions of interest and the time involved in testing at all condictions of interest.

10.5.4.3 Vehicle Structural Dynamics Simulation

The structural dynamics studies conducted as a part of the Phase II effort require considerable elaboration and expansion in the FSD Phase to assure a vehicle structural arrangement free of instabilities over the complete range of rotor RPM for both ground and flight conditions. The interaction of the envelope, suspension system, interconnecting structure, control system, and rotor systems must be modeled including the possible effect of a modified or unsteady rotor environment. The output of the structural dynamics simulation may have input to the definition of the control system, the interconnecting structure, the interface of the helicopters to the gimbals, etc.

10.5.4.4 Envelope/Suspension System Analysis

The two-dimensional symmetrical loading computer analysis developed during Phase II for describing the shape of the envelope when subjected to crosswind conditions also requires elaboration and expansion during the next phase of the HLA program. Further developments required in this area include:

(1) Expansion of the current program to a two-dimensional unsymmetrical loading capability.

(2) Expansion of the current two-dimensional symmetrical loading program to a three-dimensional capability.

(3) Development of a program with a three-dimensional unsymmetrical loading capability.

These programs all require pressure distribution information for the condition under consideration. As a result, the 40 x 80-foot wind tunnel testing is instrumental in terms of achieving an accurate appraisal of the envelope/suspension system behavior. These programs will support both the definition of the envelope/suspension system and mooring system.

10.5.4.5 <u>Continued Analysis of Control Laws and Concepts</u>

Considerable additional analysis and refinement of the current control laws by means of the vehicle flight dynamics simulation is in order during the FSD Phase of the FRV program as discussed previously. As also noted earlier, the possibility of a new method of control was revealed in the recently completed testing in the ARC 7 x 10-foot facility. If additional testing proves this method plausible, its inclusion into the control laws will be evaluated by use of the flight dynamics simulation.

10.5.4.6 <u>Continued Evaluation of Mooring Concept</u>

As noted earlier in this report there is a general deficiency of aerodynamic data at large angles of side slip for airship in the presence of the ground. The recommended testing in the ARC 40 x 80-foot facility will provide the necessary data to permit the mooring concept to be finalized. If the side force coefficient at $\beta = 90°$ proves to be substantially larger than that considered in the current mooring system analysis the weight penalty incurred in the envelope group may not justify continued consideration of the center point mooring concept. Goodyear has evaluated alternative mooring concepts as discussed previously. As required, these alternative concepts would be reconsidered based on the wind tunnel results.

The FRV once available will permit final development of the mooring system which of course is similar to the historical evolution of ground handling and mooring systems.

10.5.5 Flying Model

The benefits of a flying model have and continue to be assessed by Goodyear as a needed step in the development of the FRV. A flying model offers the opportunity to combine several of the involved technologies (e.g. aerodynamics, flight dynamics, controls) but not without considerable compromise. It is not possible to preserve all of the dimensionless parameters upon which the various phenomena associated with the experiment will depend. It appears a Froude model may be most meaningful since this will permit the vehicle flight dynamics to be addressed. A Froude model, however, does not result in the Reynolds number being scaled and this may result in unacceptable Reynolds number effects. Future wind tunnel investigations will be structured to provide data that will permit an accurate assessment of the magnitude of the Reynolds number effects.

One method of resolving the above difficulty is to fly the model inside a large structure such as the Goodyear Air Dock. In such a case, the aerodynamic forces go to zero and the Reynolds number scaling problem does not exist. The only short coming of this compromise is that the response of the HLA to atmospheric turbulence is a major area of interest from a flight dynamics standpoint. Further investigations are essential into the technical worth of a flying model prior to including such an effort as a part of the FRV development plan.

10.5.6 Remaining Phases of the FRV Program

The remaining phases of the FRV development program are sufficiently straightforward that only a few comments are included here relative to these elements.

10.5.7 Pilot and Crew Training

A considerable pilot and crew (including ground crew) training program will be required with both class room and field training necessary. The initial vehicle will be piloted (both command and safety) by qualified helicopter pilots. It is envisioned that they would receive the necessary LTA flight and ground training in a Goodyear advertising airship. Ground-based simulator training will provide the basic training tool. The ground crew for the initial vehicle would be experienced ground handling personnel currently available within the Goodyear organization. These personnel would require some training with respect to the helicopter aspects of the vehicle.

10.5.8 Assembly and Erection

As illustrated earlier in this report (see Figure 5.1) the Phase II configuration,which has been recommended as a point of departure for the FRV program, can be assembled and erected within existing facilities. The erection of the envelope for this vehicle should present no significant problem and generally will be accomplished substantially similar to prior non-rigid ariships. As indicated in Figure 10.4, within 44 months from program go-ahead it is anticipated that a research vehicle could be readied for ground and flight testing.

10.5.9 Ground and Flight Testing

Following the vehicle erection, extensive ground testing including subsystem checkout will be required.

The major purpose of the ground testing effort, of course, is to confirm that the vehicle is ready for flight testing. Included in the list of ground tests that do not fall under the category of checkout is the static and dynamic evaluation of the interconnecting structure. The static evaluation would include the application of a series of limit load conditions to the structure with the resulting stresses throughout the structure measured using the instrumentation installed for flight research. Both a natural frequency survey with appropriate dwell periods at resonance conditions and random vibration testing will be necessary. The research instrumentation would also be monitored throughout the resonance and random vibration testing.

The ground testing effort would also include, after subsystem checkouts, limited docking, undocking, ground handling, mooring and taxi tests. These tests would be in preparation for the flight test program and as such would permit flight program personnel an opportunity to gain the necessary familiarization with the vehicle prior to initiation of the flight test program.

The major purpose of the FRV flight test program illustrated in Figure 10.4 is to verify that the vehicle is ready for the intended research flight activities. In general, this flight test program will include as a minimum a matrix of tests encompassing the conditions over which the vehicle will be used in its flight research role.

Other flight tests that would be performed that would generally attest to the capability of the vehicle to adequately respond to in-flight anomalies would include rapid ascent and rapid descent as well as rapid ascent through pressure height.

10.6 Proof of Concept Flights

Following the flight test program indicated in Figure 10.1, the FRV would be ready for flight research and proof of concept purposes. Although not indicated in Figure 10.1 the proof-of-concept flights might include demonstrations of key vehicle requirements under actual operational conditions. Following these proof-of-concept demonstration flights, the vehicle would serve in the intended research role with the appropriate areas listed in Figure 10.2 being the major areas of investigation. It is anticipated that such an initial research program would involve at least a six-month period. In accordance with the plan of Figure 10.1, at the completion of the initial research flight test program the vehicle would be modified to permit the promising parallel technology program results to be evaluated. Prior to discussion of the second series of flight tests involving the modified FRV the general nature of several of the recommended technology programs is presented.

It should be noted that a market survey based on the type of economic data developed in this study is necessary in order to define the cost benefit of the recommended technology programs.

10.7 Recommended Technology Programs

10.7.1 General

As noted earlier in this TAA there are technology programs which, if successful, will contribute significantly toward improved economics, safety, and performance of this class of heavy lift vehicles. It is recommended that the technology effort associated with the HLA include those programs listed in Figure 10.1. These programs should be initiated essentially coincidental with the FRV program if the advances eminating from the programs are to be available for inclusion on the FRV following the initial series of reserach flights. There are many of these recommended technology programs that have spin-off benefit to other LTA vehicles currently under study. It is assumed all future LTA technology efforts would reflect due consideration of all requirements of these different types of LTA vehicles.

10.7.2 Fabrics Technology Program

A technology program oriented at non-rigid and rigid airship fabrics can potentially have substantial economic and safety benefits.

As noted earlier in the report, vehicles beyond the 300-ton payload capacity range will require a hull volume requiring seam strengths generally regarded beyond what can be attained in the foreseeable future. It is likely that such large hulls would employ a rigid airship type structure. As a result, the fabric technology program should be inclusive from the standpoint that non-rigid envelope and ballonet fabric and rigid envelope and gas cell fabric should be considered.

The performance benefits (i.e. reduction in empty weight and envelope volume) are of less significance than potential economic and safety benefits. For instance, in the case of the Phase II heavy lift configuration if the total fabric weight were reduced 50% (which is probably not possible given even the most favorable combination of advanced technologies), the required envelope volume is reduced from 2.5×10^6 to 2.25×10^6 cubic feet. This is still a large hull with essentially the same mooring and ground handling characteristics and the same hull-rotor interference characteristics.

Major benefits to be derived from the inclusion of advanced materials technologies is lower manufacturing costs and enhanced life charact ristics with reduced maintenance. In addition the possibility of including a rip-stop provision in non-rigi' envelopes should be investigated. This latter suggestion is tentially of benefit from the standpoint of protection of the envelope from destruction during docking and undocking operations.

There are many considerations far beyond the scope of this effort involved in the definition of the fabrics technology program. In general, however, the potential benefits offered by films, film fabrics, triaxial weaves, Kevlar, rip-stop designs, elastomers permitting heat sealed seams, etc. need to be defined. A well devised specimen test program coupled with appropriate analytical efforts can be expected to provide sufficient data to permit the benefits offered by the above to be assessed.

Table 10.2 illustrates the specimen level fabric qualification tests that new airship fabrics of the neoprene coated polyester construction are subjected to. It should be emphasized, however, that new fibers, new fabrics, etc. may have unique characteristics requiring other tests before their suitability to all apsects of airship application is assured. Final assurance that a fabric involving a new element (e.g. a triaxial weave substrate) is suited to all apsects of the airship application requires, of course, actual usage in an envelope. Careful monitoring and evaluation of specimens from the new envelope fabric over a period of several

TABLE 10.2 SUMMARY OF QUALIFICATION TESTS (NEOPRENE COATED DACRON)

Fabric End Use	Weight - Small Specimens	Strength - Strip Tensile	Strength - Cylinder Burst (Tensile)	Strength - Hot Cylinder Burst (Seam)	Strength - After Crease	Strength - After Rotoflex	Strength - Seam	Adhesion - Ply	Adhesion - Bias Ply Seam	Adhesion - Coat	Diffusion - Original	Diffusion - After Rotoflex	Diffusion - After Hot Cyclic Load	Elongation - Cylinder Stretch	Tear Test - Center Slit	Shear Modulus - Torsion Cylinder	Ageing Test - Oven - 60 days - 158°F (70°C)	Ageing Test - Oven - 24 hrs - 250°F	Ageing Test - Cold Fold
Structural Fabrics																			
Main Envelope	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Rip Panel	X	X	X		X	X		X		X	X	X			X				
Ballonet	X	X			X	X	X	X		X	X	X			X		X	X	X
Airline & Sleeve	X	X			X	X	X	X		X	X	X			X				
Inside Catenary	X	X			X	X	X	X		X				X	X		X	X	X
Outside Catenary	X	X			X	X	X	X		X				X	X		X	X	X
Patches	X	X						X		X									
Reinforcements	X	X						X		X									

years would be required. The FRV, of course, will provide this opportunity. It may not be necessary on an initial screening basis to perform all the evaluations indicated in Table 10.2. Each candidate will require careful consideration, however, to define those tests that are required with some fabric constructions perhaps requiring tests not included in Table 10.2.

During the consideration of Kevlar, whether in a coated plain weave fabric, a film laminate, or other form, it may be necessary to consider: (1) basic yarn design; (2) yarn finish; (3) type of Kevlar (49 or 29); (4) yarn treatment for proper adhesion to the elastomer or film and proper self-abrasion protection; (5) weave design to permit proper seam strength development and to minimize self-abrasion, etc. In the case of heat sealed seams the problem of quality control must be carefully addressed because these structures must be man-rated.

The qualification of any fabric, including current airship types, must address the unique conditions (by comparison to prior airships) that the HLA design introduces. Thus, the summary of qualification tests of Table 10.2 may well require expansion for this reason also. An example of the type of additional testing that may be required, is in the area of the suspension system. The suspension system in prior airships have, to a large extent, been loaded under essentially static conditions (i.e. car weight, etc.). The suspension system of the HLA will experience very dynamic loading conditions by comparison to prior designs as will any future non-rigid design using large vectorable propulsive forces.

There is substantial reason to believe, based on past experience with current airship type fabrics (neoprene coated Dacron), that they will perform over long periods of time in this dynamic environment without unacceptable loss in original properties. The acceptability of candidate fabrics to the unique loading conditions of the HLA will require developmental efforts in terms of specimen level evaluations. The most appropriate type of specimen level evaluation(s) would then likely become a part of the standard fabric qualification program.

Initially, the Fabrics Technology Program would begin with a compilation of possible candidates for evaluation and the development of a methodology by which each combination would be screened. The screening would conceivably involve both test and analysis with the analysis including a compilation and review of all applicable background data. Those candidates judged successful in the initial screening would be considered against a more comprehensive series of tests similar to those listed in Table 10.2 for current airship fabrics.

The candidates still appearing favorable would receive another thorough review from an overall standpoint including cost benefit. Those finally emerging as substantially more attractive than today's approach would be evaluated on the FRV by actual FRV modification. It may be necessary to carry a control specimen of today's technology along for detailed comparative purposes. Undoubtedly, the methodology by which candidate fabrics are evaluated at the component level will change and may be considerably different than what has been suggested here.

It will be necessary in terms of the candidate specimens as well as with today's neoprene coated polyester envelope fabric approach that seam strengths be increased if non-rigid designs are to be considered for configurations with payload capacities up to 300 tons. A 300-ton payload capacity HLA would involve an envelope on the order of 10 million cubic feet which is generally regarded as the largest non-rigid design attainable based on known seaming methods and materials. Larger HLA's would, as mentioned earlier, necessitate rigid airship construction approaches barring a fabric seaming technology breakthrough.

10.7.3 Rotor Systems Technology Program

As noted at the outset of this report, the Phase II HLA uses complete helicopters only as a cost reduction technique for the initial vehicle. It is clear that from a production standpoint a more cost effective configuration will be realized with the rotor systems retained at the outrigger extremities and the crew and vehicle controls, etc., contained in a control car similar to past airships. The control car would be placed along the bottom centerline of the vehicle which would also tend to enhance the pilot's ability to manually maneuver the vehicle. This approach reduces vehicle weight and acquisition cost, of course, because the redundant portions of the individual helicopters that are not required are eliminated. Appendix G provides an empty weight estimate for the operational vehicle. Additionally, operating costs are reduced since the flight crew can be reduced from a minimum of five to a maximum of three.

An approach is available using the original FRV with modification to evaluate the above configuration. The CH-54B helicopter has existing manufacturing break points that would permit the cockpit and tail sections to be demated. In order to have directional control at minimum gross weight once the aft tail rotors are removed, it will be necessary to be able to pitch the helicopters essentially 90°. It would appear through appropriate fuel, lubrication, and control system modifications that the CH-54B rotor/turbine system can operate under the required range of attitude conditions. The required range of roll attitudes for the rotor system would not change appreciably from that required in the Phase II design. Much additional analysis is required to confirm that this modification approach can be implemented.

There are several other possible approaches that require consideration in any attempt to refine the initial FRV configuration from this standpoint. Whatever approach is ultimately found to be most attractive, the FRV through appropriate modification would provide a method for ultimate evaluation and refinement.

As mentioned earlier, there is a definite need with respect to a Rotor Systems Technology Program to investigate low maintenance rotor systems that may result from the ability to provide greater design margins in the dynamic components without significant performance penalties. It is recommended at the outset of such a Rotor Systems Technology Program that all approaches for reducing acquisition and maintenance costs be evaluated from the combined aspect:

(1) Feasibility with respect to the HLA concept.

(2) Cost to develop technology and associated risk factor in implementing concept.

(3) Projected total cost benefits resulting from each concept in comparison to initial FRV configuration.

(4) Assessment and definition of peripheral benefits and deficiencies resulting from each concept.

As a result of this type of an evaluation, which might well involve several months of design and analysis effort by a substantial number of personnel, the general direction which the Rotor Systems Technology Program should proceed would be defined.

10.7.4 Snow and Ice Accumulation Technology Program

Rather extensive flight testing in the mid-1950's indicated that icing and snow accumulation in flight is not a serious problem for airships. However, when moored, heavy accumulations of snow and freezing rain must be avoided. Conventional airships are very sensitive to this problem whereas the HLA with its wide based landing gear arrangement and large suspension system load capability will be much less sensitive. However, if this vehicle is to become widely used in civil and military applications, the capability to operate in northern latitudes during the winter season without hangar facilities must be achieved. The only significant problem currently envisioned which may prevent this is snow and ice accumulation when moored. As a result, a technology program is recommended to continue the rather significant efforts underway in the late 1950's and early 1960.

The following approaches have been proposed or evaluated previously and require review and reconsideration in view of the developments in the last 15 years that may be applicable to this problem.

(1) Mechanical System
- a. Vibration
- b. Scraping and Brushing
- c. Distortion
- d. Covers

(2) Thermal Systems
- a. Superheating Helium
- b. Water
- c. External Heat

(3) Chemical Systems
- a. Surface Coatings
- b. Dispersal Systems

10.7.5 Cargo Handling System

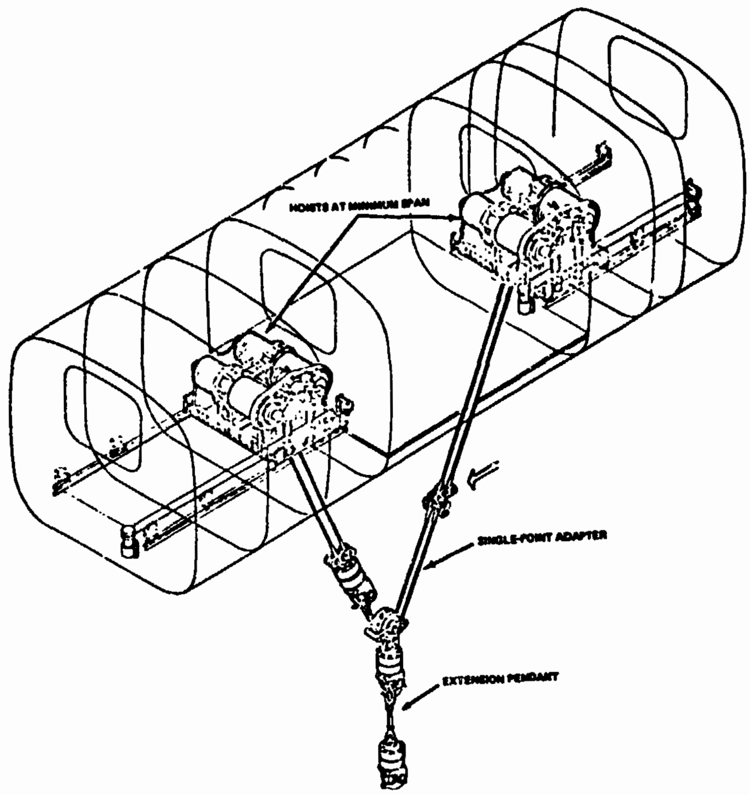
The application of the HLH precision hovering sensor to provide precise station keeping and hover maneuvering capability will likely permit a wide variety of heavy lift missions to be performed without use of any cargo handling system. The precise control of the payload can be significantly improved by cargo positioning lines operated by ground personnel during precision placement or extraction operations.

Evenutally, however, it is envisioned that missions involving extraction from confined areas will be a desired capability of this type of vehicle. Additionally, an inflight cargo attitude adjustment capability will undoubtedly emerge as a desirable feature. The pitch control power inherent in tandem rotor helicopters results in relatively little sensitivity to longitudinal c.g. displacement of the cargo. The HLA class of vehicles will be even less sensitive because of the greater distances between fore and aft rotors. The HLA will also be insensitive to lateral c.g. displacements because of the wide lateral base between rotor systems.

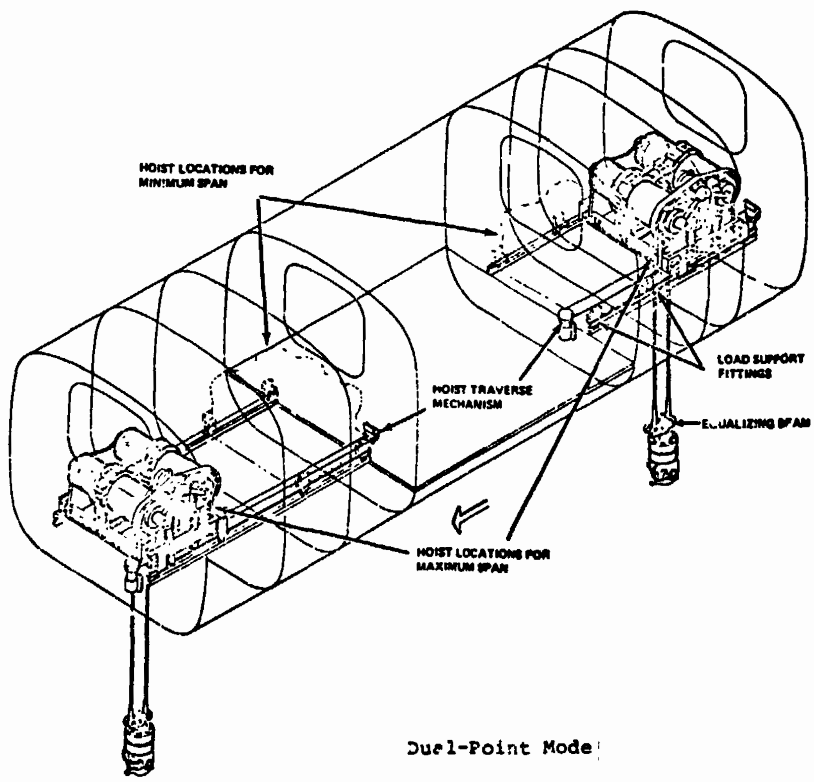
The initial effort in this technology program should be the adaptation of the existing HLH cargo handling system (see Figure 10.6) to the FRV. On a parallel basis, preliminary design studies of cargo handling systems sufficiently larger to handle projected civil and military payload requirements should be conducted. These design studies would ultimately lead to a preferable concept for development. Following the development of a cargo handling system for handling very large payloads the system should be evaluated during the second series of research flights. While the 75-ton payload capacity of the FRV would prohibit a complete evaluation of a very large cargo handling system, it would generally atest to the workability of the design.

11.0 CONCLUSIONS AND RECOMMENDATIONS

Significant conclusions resulting from the investigation of the heavy lift airship concept are:



Single-Point Mode



Dual-Point Mode

Figure 10.6 HLH Cargo Handling System General Arrangement

(1) The concept, which combines buoyant lift derived from a conventional helium-filled non-rigid airship hull with propulsive lift derived from conventional helicopter rotors, appears to be technically feasible and has the potential for meeting a growing national need in the heavy and very heavy vertical lift of large outsized cargo.

(2) The buoyancy, in addition to permitting a quantum increase in single vehicle vertical lift capability, provides a significant reduction in current vertical lift costs. Additionally, the buoyancy reduces the fuel requirements for lifting and transporting cargo in comparison to current helicopter systems.

(3) The exploratory wind tunnel investigation indicates the basic feasibility of the Phase II configuration with no appreciable interference effects for angles of sideslip up to 60° ($\alpha = 0°$). At 90° of sideslip ($\alpha = 0°$), the rotors induce a flow condition over the hull which decreases the crosswind velocity in which station can be maintained. The testing further indicated that practical modifications to the vehicle can significantly improve the observed interference effects.

(4) A six degree of freedom flight dynamics simulation has been developed and used to establish that adequate vehicle control can be achieved from the available rotor thrust forces and that good vehicle response can be expected in the precision hover mode during gusty conditions. This conclusion eliminates a major deficiency of past airships which did not possess low speed control capability.

(5) The static heaviness of the HLA combined with the available rotor thrust can significantly minimize airship ground handling problems and personnel requirements.

(6) The technology assessment analysis has indicated that a flight research vehicle is required to support the acquisition of technical information needed in the development of HLA vehicles meeting current and projected civil and military heavy lift needs.

Such a vehicle is a requirement to obtain research capabilities that cannot be duplicated in ground-based facilities or in ground-based component and subsystem testing. In addition, this vehicle will:

(a) serve a concept verification function,

(b) provide a means to illustrate advances afforded by new technology,

(c) serve to establish potential user confidence, and

(d) illustrate economic competitiveness.

The flight research vehicle would maximize use of existing government assets to minimize costs without compromising research capabilities.

(7) The technology assessment analysis has indicated that successful technology programs will contribute significantly toward improved economics, safety, and performance of the size of vehicle investigated during the Phase II study and toward larger vehicles that are projected for future civil and military needs.

Significant recommendations resulting from the investigation of the HLA concept are:

(1) In the near future, the development of a flight research vehicle should be undertaken for the reasons delineated above. The initial phase of this program should be a comprehensive engineering

program oriented at the acquisition of sufficient analytical tools and empirical data to permit the development of a minimum risk configuration possessing the needed research capabilities.

(2) A series of parallel technology programs aimed at improved performance and economics should be pursued. These would eventually be evaluated on the flight research vehicle.

(3) A market study is required to better define commercial market size and the type of vehicle and mission parameters for which to optimize future vehicles.

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